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Advanced Transportation System Studies

Technical Area 3

Alternate Propulsion Subsystem Concepts
NAS8-39210
DCN 1-1-PP-02147

Final Report
DR–4
Volume I – Executive Summary

April 2000

Prepared for
NASA Marshall Space Flight Center

The Boeing Company
Rocketdyne
6633 Canoga Avenue
Canoga Park, California 91303

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Introduction

This is Executive Summary of the Final Report covering the efforts under a NASA NRA – NAS8-39210, Advanced Transportation Systems Studies, Technical Area 3 (TA3), Alternate Propulsion Subsystem Concepts. There are three other Technical Areas contracted under the NRA. TA3 is managed through MSFC/PD with Gary Johnson as project manager. The contractor team is led by Rocketdyne with Thiokol, Workingsolutionz Software, Davis Aerospace, and the University of Alabama as team members.

The contract started on 6 April 1992 and continued through April 2000.

The objective of the contract was to provide definition of alternate propulsion systems for both earth-to-orbit (ETO) and in-space vehicles (upper stages and space transfer vehicles). For such propulsion systems, technical data to describe performance, weight, dimensions, etc. will be provided along with programmatic information such as cost, schedule, needed facilities, etc. Advanced technology and advanced development needs will be determined and provided.

A propulsion system database was also developed which is capable of including the systems examined under TA3 and any other existing or conceptual propulsion systems.

The contract results are reported in three parts:

Volume I – Executive Summary which overviews each of the contract tasks giving its objective, main results, and conclusions;

Volume II – Final Report which references the individually delivered detailed Task reports (the detailed results are in the separate Task Reports, not in Volume II) and fulfills the requirements of a place to Report DRs 8 (Computer Aided Design Graphics and Analysis Data Documentation and Transfer) and 9 (New Technology Report), neither of which had any activity to report;

Volume III – Program Cost Estimates which contains DRs 5 (Work Breakdown Structure (WBS) and WBS Dictionary) and 6 (Program Cost Estimates Document).

Discussion

The Alternate Propulsion Subsystem Concepts contract had seven tasks defined that are reported under this contract deliverable. The tasks were: F-1A Restart Study, J-2S Restart Study, Propulsion Database Development, SSME Upper Stage Use, CERs for Liquid Propellant Rocket Engines, Advanced Low Cost Engines, and Tripropellant Comparison Study.

The two restart studies, F-1A and J-2S, generated program plans for restarting production of each engine. Special emphasis was placed on determining changes to individual parts due to obsolete materials, changes in OSHA and environmental concerns, new processes available, and any configuration changes to the engines.

The Propulsion Database Development task developed a database structure and format which is easy to use and modify while also being comprehensive in the level of detail available. The database structure included extensive engine information and allows for parametric data generation for conceptual engine concepts.

The SSME Upper Stage Use task examined the changes needed or desirable to use the SSME as an upper stage engine both in a second stage and in a translunar injection stage.

The CERs for Liquid Engines task developed qualitative parametric cost estimating relationships at the engine and major subassembly level for estimating development and production costs of chemical propulsion liquid rocket engines.

The Advanced Low Cost Engines task examined propulsion systems for SSTO applications including engine concept definition, mission analysis, trade studies, operating point selection, turbomachinery alternatives, life cycle cost, weight definition, and point design conceptual drawings and component design. The task concentrated on bipropellant engines, but also examined tripropellant engines.

The Tripropellant Comparison Study task provided an unambiguous comparison among various tripropellant implementation approaches and cycle choices, and then compared them to similarly designed bipropellant engines in the SSTO mission.

Figure 1 shows the schedule for the first year of the contract. Figure 2 shows the reviews which took place and the documentation available.

F-1A Restart Study

The NASA/Rocketdyne F-1 engine completed its production run in 1969 after delivery of 98 units, 65 of which were flown on the Saturn V launch vehicle with 100% success. Nearly 255,000 seconds of hotfire testing was accumulated on the production engines and 56 equivalent development engines during the program. Development efforts included more than four years of design, analysis and testing of an F-1A engine with the capabilities of 1800 Klb thrust and of throttling as well as reduced production and operational costs. This knowledge and experience provides the foundation for a 1990's F-1A. A comparison of the F-1 and the F-1A engines is shown in Figure 3.

Figure 4 shows the overall context in which the F-1A Restart task of this NRA was performed. It was only one part of a larger effort needed to assess the restart of the F-1A.

The F-1A Restart Program is based on a multi-phase, incrementally funded plan, which when fully executed, will provide the technical and programmatic foundation necessary to support a NASA decision on F-1A production. The initial feasibility evaluation effort was performed by Rocketdyne in 1990-1991, using discretionary resources. This effort was targeted at assessing the availability, completeness, quality and usefulness of F-1/F-1A documentation, hardware, tooling, supplier, facility, and personnel resources. This information along with mission planning analysis, customer requirements input, and Rocketdyne's recent ELV Program restart experience, was used to assess the potential effectiveness and viability of the F-1A engine in a 1990's booster application. Rocketdyne's conclusion at the completion of this effort was that a customer need did exist, and that, indeed, a sufficient "critical mass" of F-1A knowledge, experience and hardware assets was available to warrant further, more detailed investigation of the feasibility of an F-1A Production Restart Program.

Phase A of the Restart Program Plan was formulated to address, in detail, the configuration, manufacturing, and test issues associated with an F-1A production restart so that detailed program schedule and cost estimates could be developed. The effort funded in this NRA focused on that portion of Phase A that would refine the requirements for a 1990's F-1A. The remaining Phase A effort consists of two parts. The first would prepare detailed Manufacturing and Test Plans, and prepare refined program cost and schedule estimates. The second part is an effort in which Rocketdyne would support the return, disassembly, and evaluation, at MSFC, of an F-1 resource engine.

Phase B of the program would focus on selected technology demonstrations, hardware assembly efforts, and ultimately an engine hotfire demonstration test.

The study objective of the NRA task was to determine what is realistically required to take the 1960's F-1A engine, produced at the end of the F-1 program, and bring it back into cost-effective production in the 1990's. To meet this objective, Rocketdyne established a design approach which balances the use of the existing F-1/F-1A designs with their demonstrated reliability, mission effectiveness and extensive development history with high-value recommended process improvements and design modifications which incorporate state-of-the-practice fabrication methods, producibility enhancements, and which address 1990's material substitution issues associated with environmental regulations, obsolescence, and stress corrosion susceptibility. The modifications were required to retain the form, fit, and function of the original components. The approach of the F-1A Restart Program is shown in Figure 5.

The results of the study would then provide the necessary foundation for the detailed manufacturing and test plans and non-recurring and recurring cost estimates that are needed to complete the effort as described in Figure 4, Rocketdyne's Phase A, F-1A Restart plan.

The groundrules for the F-1A Restart Study were:

Comply with identified NASA requirements:

1. capable of producing 1800 Klb thrust
2. provide a throttling capability to 75% thrust, or 1350 Klb
3. incorporate health monitoring features necessary to support engine test and launch operations.

Incorporate 1990's state-of-the-practice fabrication methods to reduce production costs and part variability

Provide recommended design modifications to address structural margin (i.e., maintain the margins of the original F-1), producibility, material substitution, and reliability/operability issues.

A set of F-1A component plans was developed for significant parts and used to establish design and process requirements for a 1990's engine. A total of 59 component plans, as shown in Figure 6, were developed, 57 individual plans and two summary plans for the turbopump and thrust chamber which summarize the detailed component plans within those assemblies.

Figure 7 shows a summary of the recommended process improvements and design modifications identified in the 59 component plans developed during this study.

The first category, the one-hundred nineteen State-of-the-Practice fabrication method improvements represent the single largest category (53%) of recommended changes for the F-1A. They also represent the most "transparent" of the recommended improvements, in that their implementation will have no effect on the configuration or qualification status of the hardware, and therefore requires nothing other than the standard first article demonstration that would be required of any process modification.

Seven categories of fabrication methods were considered: robotic operations such as welding; Numerical Control (N.C.) machining, which would be driven from CAD databases; laser operations such as drilling, trimming and welding; casting improvements such as use of investment or lost foam methods instead of sand casting; forging methods to improve material properties; machine cell operations in which similar operations could be performed on similarly sized hardware to reduce setup and queue times; and multi-purpose tooling which would be adaptable for a number of parts, or which could serve both as inspection and machining fixtures. Figure 8 summarizes the major components and the recommendation for implementation or further evaluation of new or improved fabrication methods.

The most frequently identified fabrication method improvement was use of N.C. machining. Fifty-two opportunities for implementation of this technology were identified in the 59 Component Plans. This is also significant with respect to the component design documentation, since N.C. machining instructions can readily be down-loaded from CAD databases. The F-1A Restart Study recommended that any components which can benefit from N.C. machining technology should do so using a characteristics database provided from CAD drawings.

The second category of recommended modifications was material substitution changes. Forty-two material substitution recommendations for implementation or further evaluation were identified in the F-1A Component Plans. These are summarized in Figure 9. They include fourteen modifications driven by the need to comply with regulations regarding the use of hazardous materials such as beryllium, cadmium, and asbestos; five modifications to replace obsolescent materials such as Inconel X 750 and Hastelloy C; and twenty-three evaluations of stress corrosion susceptible materials such as 321 CRES, 17-7 PH steel, 2024-T351 aluminum, and 7075 T6 aluminum.

TENS 50 aluminum, which was used to cast a number of the components on the MK10A turbopump, is an alloy which contains a percentage of beryllium which exceeds the current OSHA standard. Two recommended alternate materials have already been identified for replacement of TENS 50 depending on the required material strength of the affected component. A356 aluminum contains no beryllium, but is lower strength than the TENS 50. A356 aluminum would be used if material strength is not an issue for a given component. A357 aluminum contains beryllium, but at a percentage that complies with the OSHA regulation. This alloy, which is equivalent in strength to TENS 50, would be used as a replacement when the higher strength is required. The Rocketdyne ELV engine restart programs (Atlas and Delta engines) successfully converted turbopump impellers, inducers and volutes from TENS 50 to either A356 or A357, as appropriate. These changes were carried out with no impact on the production program, because of their similarity to TENS 50.

Cadmium and asbestos are well known hazardous materials. Alternates for cadmium plating are readily available, and will not be an issue for the F-1A. The asbestos F-1 thermal blanket will be replaced with materials successfully implemented on the RS-27 engine thermal blanket during the ELV program restart effort.

The third category of recommended modifications was producibility enhancements. These producibility enhancements were an important part of the F-1A component evaluation effort. Implementation of these kinds of changes can not only significantly reduce the cost of the hardware, but can also enhance quality by eliminating failure modes and reducing part to part variability. Rocketdyne's recent restart of the Atlas and Delta engine programs identified and successfully implemented a number of producibility enhancements. These included conversions of welded assemblies to castings, material substitutions to enhance process operations, and design simplifications to eliminate unnecessary processing. These same types of producibility hardware modifications have been identified for the F-1A.

Thirty-five component design modifications or recommendations for further evaluation were identified as producibility enhancements in the F-1A Component Plans. These include: nine conversions of welded assemblies to castings; two design simplifications to the existing LOX and Fuel Volute castings; 16 modifications to simplify designs on components such as the turbine exhaust manifold, nozzle extension, thrust chamber tubes, and ducts; and eight material changes to reduce the processing requirements of components such as the high pressure propellant ducts and gimbal bearing.

An example of the producibility enhancements is converting welded assemblies to castings. Nine of these were recommended as shown in Figure 10: the two MK10A fuel inlets; the MK10A turbine manifold; the thrust chamber LOX dome, fuel manifold and jacket; the gas generator injector and chamber; and the Interface Panel. Conversion of these assemblies to castings will save approximately 80% to 90% of the labor associated with their current processing methods. For example, the LOX Dome current design contains approximately 60 details such as shell segments, flanges, pins, bosses, spacers, brackets, and a body which must be machined and/or formed. Many of the details must then be welded together, which involves joint preparation, fit-up, welding, inspection, and rework as necessary. Conversion of this assembly to a single piece casting will eliminate all welding and related preparation, and weld inspection. In addition, casting will allow the elimination of much of the machining operations, and allow the consolidation of the machining operations that are necessary into a flexible machining cell environment.

Casting technology advances, and recent successful casting conversion experience on the ELV engine programs provide confidence that these modifications can be implemented with low technical, cost and schedule risk.

The fourth category of recommended modifications was those necessary to maintain the original F-1 structural margins at the increased thrust and pressure associated with the F-1A thrust level.

F-1A engine performance models were used to simulate 1800 Klb thrust operation. The calculated operating conditions throughout the engine were used to perform a comparative structural analysis of the components. As a result of this analysis, twenty-one recommendations for modification were made in order to maintain the structural factors of safety of the components under the increased loading associated with operation at the 1800 Klb thrust level. Components to be modified are shown in Figure 11 and include the thrust chamber; nozzle extension; heat exchanger; GG injector, chamber, propellant valve and ducts; the high pressure propellant ducts; the Main Oxidizer Valves, and the thermal protection system. Design modifications to accommodate the 1800 Klb loading will be implemented such that the original design form, fit and function are unaffected.

Many of the components to be modified, such as the LOX dome, injector, thrust chamber components, nozzle extension and gas generator chamber for example, also have recommendations for modifications in other categories. This will then permit simultaneous solutions for the multiple design modification drivers. Several of the components for example are recommended for casting conversion, and therefore, strengthening may only require minor changes to the wall thicknesses. The nozzle extension strengthening will simply be incorporated into the requirements for the selected design option.

Similar strengthening design modifications were successfully implemented on hardware such as the thrust chambers, gas generator, ducts and heat exchangers during the ELV engine production restart programs.

The F-1A Structural Margin design modifications have already been assumed in the previously provided cost and schedule estimates, and therefore will impose no unplanned impact on the design or verification test effort.

The last category of recommended modifications was a small set of changes to improve the operations and reliability of the F-1A design. The F-1A Restart Study included an overview of the component history via reliability records and interviews with personnel involved in the field test and launch operations. This review identified seven recommendations for component Reliability/Operability driven modifications. The components were: the nozzle extension; thermal protection system; the MK10A lubrication system; the GG Ball Valve lipseal, and the Main Fuel Valve Poppet seal. All of these, with the exception of the MK10A turbopump lubrication system, experienced failures and repair/replacement actions which the F-1A team has concluded should be corrected during a program restart. A MK10A turbopump with integrally cast lubrication passages was successfully tested during the F-1 R&D program. This modification allows the elimination of the Bearing Coolant Control Valve, and all of the associated lube line and valve control plumbing.

The Restart Study specifically addressed the topic of the combustion stability history of the F-1/F-1A engine. A thorough review of the engines combustion stability analysis, design and test history was conducted as a part of the Restart Study. A paper on the subject ("Combustion Stability and Large Liquid Rocket Engines – The F-1 Story") was prepared and presented at the Penn State Propulsion Engineering Research Center 4th Annual Symposium at MSFC, Huntsville, Alabama, on 9

September 1992. That paper is available as part of the F-1A Restart Study Final Report (as Appendix B).

Early in the F-1 development program, the injectors were found to exhibit unstable operation, which resulted in hardware damage. A major effort was mounted at that time to investigate and correct that condition. An injector development program designed and evaluated a number of design fixes, which ultimately resulted in what was called the "Qual II" injector. This injector was thoroughly tested, and demonstrated its stable performance during 1112 tests on 94 units. The tests included 22 bomb tests in which artificial instabilities were induced. The test results demonstrated that bomb induced instabilities were damped within 45 msec. (<100 msec. required) and that there were no self-induced instabilities. This injector configuration then became the production design and was used successfully throughout the program on the F-1/F-1A engines.

Component limits testing was conducted to demonstrate performance margins and durability during the F-1/F-1A program. These tests demonstrated satisfactory results at thrust levels ranging from 1250 Klb to over 1800 Klb. Sixty-two tests, including 50 bomb tests (9 >1800 Klb) were conducted. No self induced instabilities were noted, and all bomb induced instabilities damped within 45 msec.

F-1A R&D testing also demonstrated satisfactory combustion stability during 25 tests and 1800 sec. at 1800 Klb thrust on two engines (104-4, and 109-4)

The non-recurring and recurring cost estimates generated during the previous F-1A Restart Program (Figure 4) were examined at the completion of this study to determine if any changes were appropriate based on the study results. The study findings indicated that there were no program activities overlooked that would adversely affect the cost estimates, and that those cost elements that were included were properly estimated based on the top down estimating approach used. The elements comprising total engine cost for contractor and government are indicated on Figure 12. The costs are based on a five-engine development/certification program and delivery of 72 flight engines produced at the Rocketdyne Canoga facility. The TBD costs depend on the type of contract, the location and number of engine and component test facilities, the stage testing requirements, and the degree of Rocketdyne flight support involvement.

The study also identified a number of yet to be quantified net cost reduction opportunities. The remainder of the Phase A Restart Plan calls for the preparation of detailed Manufacturing and Test Plans which will enable the refinement of the non-recurring and recurring cost estimates for the restart of the F-1A program.

The F-1A Restart Study achieved its objective of defining the requirements for manufacturing a cost effective 1990's F-1A:

- A demonstrated F-1A configuration baseline has been established which meets all identified customer requirements. Fifty-nine component plans have been prepared identifying the 1990's F-1A design and process requirements.

- No significant technical issues were identified. The technical effort content of the F-1A Restart Program has been evaluated. Rocketdyne has concluded that it presents no significant technical, cost or schedule risk and that it has been properly accounted for in the previous recurring and non-recurring cost estimates.

- High value, low risk process improvements and design modifications have been identified. These include: state-of-the-practice fabrication methods, cost reducing producibility modifications, and regulatory compliance modifications.

Documentation. The F-1A Restart Study has been documented in three forms. There was an Executive Overview presented as part of the 1-2 October Review . There was also a separate, and much more detailed, Final Task Report submitted in October 1992. Besides the two reports, there was a computerized database of the F-1A component plans generated. It was delivered in two forms: as a 4th Dimension file which included the text fields and the graphic fields, and as FileMaker files (both FileMaker 1.5 and FileMaker Pro) which included only the text fields.

All the documentation is available from MSFC/PD.

J-2S Restart Study

The objectives of this study were to assess what design changes would be required to reinitiate production of the J-2S engine for use as a large high energy upper stage engine, as it was designed for, or the possible use as a boost stage engine. The study assessed design changes required to perform per the J-2S model specification, manufacturing changes required due to obsolescence or improvements in state-of-the-practice, availability issues for supplier provided items, and provided cost and schedule estimates for this configuration.

The results of the study provided the necessary foundation for the detailed manufacturing and test plans and non-recurring and recurring cost estimates that are needed to complete the effort to reinitiate production of the J-2S engine system.

The J-2S (J-2 Simplified) engine was originally developed as a follow-on configuration for the J-2 Saturn vehicle upper stage engine. The intent of the design was to not only provide performance upgrades to the engine but to greatly simplify the production and operation of the engine. The original J-2S effort used the same design and development team as the J-2.

The nominal vacuum thrust of the engine was 265,000 pounds while providing a specific impulse of 436 seconds with a 40:1 nozzle expansion ratio. Baseline operation was at a mixture ratio of 5.5, oxidizer to fuel, with the capability to operate at mixture ratios of 5.0 and 4.5 upon command for optimized propellant utilization during the mission. All engine interfaces were located such that the engine could be used as a direct substitute for the J-2 engine. The engine cycle was changed to a tap-off cycle to eliminate the gas generator. Throttling capability was added as an option for applications other than the Saturn Program. The engine also included a feature for low thrust operation known as "Idle Mode" which was to be used for propellant tank settling, on-orbit maneuvering, and rapid engine chilldown prior to firing.

This engine system was validated with 6 flight configuration engines in 273 tests for a total operating experience of 30,858 seconds. Upon the termination of the J-2S program, the engine was ready to go into certification for flight operations. Figure 13 summarizes the J-2S engine.

This NRA J-2S Restart task retained the previous thrust level of the engine at 265,000 pounds of vacuum thrust, even though analytical effort has suggested that the engine is capable of being easily uprated to over 320,000 pounds of thrust. Propellant utilization control for real time operation at mixture ratios of 5.5, 5.0, and 4.5 was also retained. Two versions of the J-2S were configured. One for the S-II stage with single start capability and one for the S-IVB stage with three start capability. This study examined both of these configurations and focused primarily on the three start configuration since it provided the greatest operating flexibility for possible applications. The idle mode operating capability was retained as well since the features of propellant settling, orbital maneuvering, and rapid engine chill would justify the added complexity on most applications.

Just as in the F-1A Restart Study, this NRA task also baselined the use of low risk design and process improvements to take advantage of 1990's technology. The major area of emphasis was in the area of producibility, since the J-2S was only manufactured as a development engine and therefore not ever mass produced. An area identified as a potential problem was the acceptability of some materials and processes used during the 1960's which are no longer environmentally acceptable. The J-2S manufacturing process was reviewed for any required environmental changes but none were found. Also during the study, the reliability and operability of components were reviewed to assess if any of the problems seen during the J-2S test series could be easily solved using experience gained from subsequent programs. All identified improvements were incorporated into the engine baseline.

At the outset of the study improvement groundrules were defined. These were imposed to ensure that no proposed changes would invalidate the extensive development effort already invested in the engine configuration. Only previously demonstrated cost and cycle time reduction techniques were allowed such as incorporation of castings or numerical control machining. Technologies which had been applied at Rocketdyne were acceptable but others were not, since the scope of the study was limiting and validation of any other process technology was not possible. High technical maturity was sought in the changes to assure lengthy process development would not be required. Only those processes with minimal qualification requirements were acceptable since it was considered important to keep restart cost and schedule to a minimum. It was a groundrule that original form, fit, and function were to be retained for engine components with the only exception being the engine interface itself, which could be modified for any specific application. Original structural margins were to be retained since the potential application is undefined at the moment and it was thought to be desirable to retain the engine's "man-rating". All development concerns of each of the components was to be addressed so that the restart program would not be initiated with any known problems.

Figure 14 shows the task flow used for the study.

The initial task in the J-2S restart study was to review the J-2S configuration, the engine's history, and available documentation. Documents were organized to provide a central knowledge library for use throughout the study. Personnel who participated on the original J-2S program were consulted on numerous occasions throughout the study.

After historical documents were reviewed, each of the component specialists examined their areas to propose design modifications to address producibility material substitution from either obsolescence or environmental need, reliability/operability issues, and to insert processes that are current state of the practice. Each of the proposed modifications were evaluated on the bases of being low risk while providing high added value to the engine system.

The baseline configuration was then used to define a program plan and cost estimate for the restart program. All testing and costs are consistent with the configuration defined during the producibility program.

In parallel, a 10:1 throttling examination was conducted to identify changes required to perform a large throttle ratio.

As part of the review of each of the major components, the responsible engineers and their teams reviewed the drawings to assure full understanding of how the part was previously made along with an examination of internal letters referring to operational and production issues with each of the parts. For many of the components interviews were conducted with personnel actually involved in the design, production and development of the hardware to obtain first hand feedback of the shortcomings and lessons learned from experience. In several cases retired personnel were also used to assist in these studies.

Three areas were selected for greater depth of study since documents either indicated greater concern for these parts at the end of the development program or less depth of documentation was actually available in the area. Turbomachinery was examined since this is an area of traditional concern in rocket engines and there was some record of early fuel turbopump issues seen in the historical documentation. The thrust chamber tap-off port was also examined in greater depth since the J-2S is the only combustion gas tap-off cycle ever tested and early program records indicated problems with erosion. It was felt that any issues with this part had to be fully understood prior to reinitiating production. Finally the engine valves were examined in greater depth since several operational issues were identified along with the fact that several of the drawings were missing from the storage locations.

Along with these "in-depth" examinations every major component of the engine was evaluated by the team to determine a baseline configuration. Figure 15 shows the engine components which were evaluated.

The turbomachinery configuration was reviewed. The Mk-29 turbopumps were used after the J-2S Program on the Linear Aerospike Engine program. Also the Mk-29F has recently been disassembled, modified, and rebuilt under Rocketdyne's Mk-29FD IR&D Program. This was an opportunity to take recent pump build experience and recent producibility effort done under the NLS Program to form a solid configuration baseline with personnel very experienced in this particular hardware and working on a similar producibility improvement goal.

The thrust chamber tap-off port was both a durability and producibility concern. Documents and personnel recollection indicated that the previous port configuration was very difficult and time consuming to produce. Also records indicated that this configuration had life limiting erosion issues early in the test program. This was the opportunity to solve both of these issues by introducing modern producibility techniques where applicable along with a configuration that could be designed for durability using modern thermal modeling techniques. A modified configuration was designed as shown in Figure 16 which replaces the intricately machined and hand brazed part used during previous testing, greatly simplifying the procedure and providing far greater confidence in production without nonconformances. It should also be noted that the advanced materials used far exceed the capabilities of the 321 CRES material used in the previous work, thus supplying an engine with improved operating margins along with producibility improvements.

Drawings for several of the valves could not be located and several documents indicated that problems were experienced during the development program. To address these issues a retired Rocketdyne valve specialist was brought in to provide valuable historical perspective on this task of defining a restart baseline. All previous test problems were able to be identified and state-of-the-practice solutions found. In the case of the valves whose drawings had not been found, it was determined that new drawings would be needed to incorporate improvements for operation and producibility anyway.

Several recommendations appeared to be universal in the study. The incorporation of robotics and numerical control machining to replace previously manual operations would be implemented. This provides lower part cost and greater part-to-part repeatability. The use of laser measurement and inspection techniques allows greater precision of flaw detection. Replacing simple geometry forged and welded parts with net shape castings and forgings greatly reduces the process flow of long lead parts. The use of machine cells and multi purpose tooling will eliminate the queuing times and minimize the overall tooling costs. Adopting precision end point tooling for ducts where the J-2S was previously built with custom fitted ducting due to its low production rate will decrease the process flow time and needed labor.

It was also found that all of the electronic parts required updating simply because the formerly used parts are no longer available. New electronic parts have better performance and cost much less, aside from the fact that they are available.

An interesting conclusion of the study was that the J-2S could be produced entirely from existing drawings, with the exception of replacing the obsolete, outdated, and unavailable electronics. This is a sharp contrast from other restart efforts conducted at Rocketdyne. All of the materials used are still available for use, as are the manufacturing processes.

While the engine could be produced using the existing prints, a number of low risk, high payoff changes were identified. Twenty four component changes to aid in producibility were identified that do not alter form, fit, or function. In addition to this, another twenty changes in fabrication technique were identified such as the use of modern castings or die forgings to produce previously labor intensive components. The use of fabrication techniques, along with the existence of known superior materials, yield a recommendation to perform 12 material substitutions. Finally, eleven reliability or operability enhancing changes were identified. Such changes incorporate now proven technologies that were not available during the time of J-2S design.

Several complete component redesigns were identified which would be defined as a function of intended application. The electrical control and sensor architecture would be modified to incorporate condition monitoring and health monitoring capability appropriate to the intended application. Also the propellant inlet scissor ducts would be replaced with much less costly wrap-around ducts if the interfaces of the application allow it. Since this engine has the potential of operating in applications requiring long life, the hot gas check valve would be redesigned to provide improved life margin over those units tested, although this is not necessarily required for an expendable application. Engine interfaces would also be adjusted to provide a more efficient design than that used in previous testing.

Figure 17 summarizes all of the recommended modifications by major component.

The restart study examined what modifications would be required to throttle the J-2S engine to 10:1. The study also defined what impacts would result from these modifications.

At the outset of this study, the data taken on engine J-115 at AEDC on throttling tests down to 6:1 throttling levels was examined to determine if satisfactory operation had been observed to that level. Then this data was compared to the off-design code J-2S engine power balance model to determine the accuracy of the model. Throttling cases were run at several points (1.7:1, 3:1, 5:1, 6:1, and 10;1) to establish operating trends.

It was found that the model agreed very well with the test data all the way to the 6:1 previously tested. In throttling to 10:1 the oxidizer injector element pressure drop to chamber pressure ratio dropped to below minimum stability criteria. If such deep throttling were desired, it would be possible to increase the oxidizer element orifice resistance to allow stable operation at the low power level. This would require an oxidizer orifice resistance increase. If the turbine tapoff flow were held constant, then the chamber pressure would be reduced to approximately 1000 psi. Alternately, the tapoff flow could be increased and the pump discharge pressure increased to maintain the current chamber pressure.

Turbomachinery performance at the 10:1 operating level appears acceptable based on the 6:1 experience, however this study did not address the secondary coolant and seal flows which may still require some additional modification. These results have a high degree of confidence due to the previous test data and its excellent correlation with the engine model.

This study provided the confidence that J-2S production could be reinitiated within reasonable costs and schedules. No significant technical issues were identified in either the producibility study or in the review of previous technical data. Areas of potential cost reduction were identified which could be quantified to a greater extent with further manufacturing planning. The proposed schedule can be met with no foreseeable impacts.

All of the design changes are sound technically and thus prudent risks. Many of the processes examined have already been applied to restarts of Expendable Launch Vehicle engines such as those used on Atlas and Delta launch vehicles. The changes provide reductions in cost, schedule, operability concerns, reliability concerns, or, as in the case of many of the changes, address all of these issues. Finally the testing required to fully validate the proposed changes is completely within the scope of the test series, which would probably be applicable even if no changes were made form the original J-2S drawings.

Figure 18 summarizes the overall changes suggested for the restart of the J-2S engine.

Program planning, using the results of the producibility study, was performed. A conservative engine development test plan, which can examine all pertinent operating points, was produced using four development engines and two qualification/certification engines. This plan presumed that either an altitude simulation facility, similar to that previously used at AEDC, or a diffuser nozzle was available for the test program. The total tests planned was 210 tests for a total duration of approximately 25,000 seconds.

Four of the six engines would be tested to the model specification life of 3,750 seconds while two would undergo extended testing to 5,000 seconds. This is only a preliminary test plan which takes a very conservative approach to verifying the flight readiness of the engine.

Costs of a J-2S restart program, both recurring and non-recurring, were estimated and are shown in Figure 19. For these estimates, it was assumed that the engine life requirement would be the same as the original J-2S model specification calling for 30 starts and 3,750 seconds of operation. It was also assumed that in-flight restarts would be a requirement so the engine is configured for three starts on a mission. The planning assumed that government facilities would be used wherever they were available and cost effective. A limitation placed on this planning was to limit certification to single engine configurations so that this work would not be configuration dependent. This means that additional effort would be required for clustered applications since nozzle thermal protection and main propulsion test article testing were not included. For the purpose of cost estimating, the use of Rocketdyne facilities and engine assembly were presumed which did not account for any gains to be had in co-locating production and test facilities. The planning used for production restart assumed that the existing drawings and specifications would be updated rather than transferring the drawings and specifications to electronics based systems.

Modifications to Rocketdyne facilities have been identified and estimated for areas where such testing would occur. Finally, the cost of the propellants were not included in the estimates since this is highly dependent on facility configuration, test program, and test location.

Documentation. A separate Final Task Report has been submitted to MSFC/PD. It contains a detailed listing of the producibility study, details of the turbopump and tap-off port studies, and program development plans. A shorter overview is contained in the briefing book for the 17 March 1993 Final Program Review. It is also obtainable from MSFC/PD.

Propulsion Database Development

The objective of the database development task was to produce a propulsion database which is easy to use and modify while also being comprehensive in the level of detail available. The database was to be available on the Macintosh computer system. This task extends across all three years of the contract. Consequently, a significant fraction of the effort in this first year of the task was devoted to the development of the database structure to ensure a robust base for the following years' efforts. Nonetheless, significant point design propulsion system descriptions and parametric models were also produced.

It is desirable that the database be usable for both the preliminary analysis of whole classes of propulsion systems (e.g., a booster engine using LOX/RP for a wide range of thrust levels) and for the analysis of existing propulsion systems (e.g., SSME, RD-170, etc.). Since it would be very difficult to fulfill both these uses with only one database structure, it was decided to develop two separate tools, one for each type of usage.

The first usage (analysis of classes of propulsion systems) is normally implemented by a series of unrelated tools written as spreadsheet models, or as dedicated code (most commonly written in Fortran) and running on mainframes, workstations, or PCs. These tools normally can not communicate with each other and are written without common structure – they calculate weight breakdowns to different sets of components even for similar engine types and calculate performance in different manners. This usage requires large amounts of calculations, methods of data presentation unique to each propulsion type (and sometimes to different engine classes within a type), and benefits from automated parametric data generation and automated preparation of graphs (e.g., weight versus mixture ratio).

The commercial tool type which comes closest to meeting these needs is a spreadsheet, particularly one with good graphing capabilities, an extensive scripting or macro language, and the ability to access external code written in different computer languages (especially Fortran). Both Resolve and Excel were considered and Resolve was chosen because its scripting language is extensive and very easy to use even by casual users, and because its charting capabilities (including the scripting of all elements of each chart) were more extensive than Excel (at least until Excel 4 which was not available to the author at the time). It subsequently became known that Resolve also puts fewer limits on the use of Fortran externals than Excel. This first usage type will be referred to throughout the rest of the report as a "parametric propulsion database".

The second usage can be implemented with a classic database structure where a large number of pieces of information (as numbers, text blocks, and pictures/graphics) about each of a number of specific existing or conceptual propulsion systems is stored. The information describes the single design point engine with some information about operation at off-design conditions. Each propulsion system can be stored as a record with the individual pieces of information stored as fields within the record. Minimal calculation is needed, but the ability to sort, group, and aggregate (i.e., all engines using RP with vacuum thrust above a specified number) is needed. Consequently, for this usage, referred to throughout the rest of the report as a "propulsion system database" a commercial database was chosen. Both 4th Dimension and FileMaker Pro were considered. FileMaker Pro was chosen because it is much easier to change, both in structure and output, even by casual users. It is also much more readily available because of its much lower cost, cross platform capability (Macintosh and PC with Windows), and lack of need of dedicated, experienced users.

Parametric Propulsion Database. The parametric propulsion database was developed using the Macintosh spreadsheet Resolve, version 1.1v1 (published by Claris). It was developed on a Macintosh II fx running system 7 with the tuneup kit. It was developed using an Apple 13 inch color monitor. It has been checked in black and white mode, on a limited number of other Macintosh computer types, and with system 6.0.5. Two problems were encountered during these checks: some color choices were changed to work in black and white mode, and the Fortran externals were recompiled in two forms so they would work on Macintoshs without math coprocessors, but would also take advantage of the coprocessors when present.

The parametric propulsion database consists of two files and one folder (which in turn contains three files):

Parametric Database
Library
Externals

The file "Library" and the folder "Externals" must be in the same folder as the application "Claris Resolve". The file "Parametric Database" can be placed anywhere. None of these file or folder names can be changed because they are used explicitly by name in calls by scripts in the database. The file "Parametric Database" is a Resolve spreadsheet which is double-clicked to run the parametric propulsion database. It uses the file "Library" to update its worksheet script. "Library" contains a number of functions which are called by other scripts. The file "Library" is actually only needed when changes are made to the worksheet script. The program will run without "Library" (although two error messages will occur) but changes cannot be made, even temporarily, to the worksheet script. The folder "Externals" contains the three compiled Fortran codes (with embedded hooks written in C – see Appendix) currently used by the database.

The model requires the fonts "Bookman", "New Century Schoolbook", and "Helvetica" be installed (Postscript or True Type). If they are not available then most screens and output will be difficult to read and many words will not be fully visible in their defined columns. All three of these fonts came with the various Apple LaserWriters (and many other printers) and are readily available. The use of Adobe Type Manager (ATM) or True Type (with the True Type versions of the fonts) is highly recommended to improve the readability of the screen.

To run the database simply double-click on the file "Parametric Database". The current version (version 1.4, 5 April 1993) contains the following models:

- Solid Fuel Boosters
  - Large Motors (328K-8.9M lbf) using ASRM (ANB3652) propellant
  - Large Motors (328K-8.9M lbf) using neutralized Mg (DL-H435) propellant
  - Medium Motors (62K-328K lbf) using neutralized Mg (DL-H435) propellant
  - Large Motors (328K-8.9M lbf) using non-chlorine (PGN/AN/AL) propellant
- Hybrid Boosters
  - Large Motor (380K-21M lbf) using O_2 as oxidizer and HTPB and escorez as fuel – pressure fed
- Cryogenic Engines
  - Large (100k-2M lbf) LOX/H_2 engines using staged combustion cycles
- Hydrocarbon Engines
  - Large (500K-3M lbf) LOX/RP engines using gas generator cycles

The solid fuel rocket booster and hybrid booster models are implemented as spreadsheet models, while the liquid engines are implemented as Fortran external functions.

The basic philosophy of the model is to navigate a large spreadsheet by means of buttons that the user "clicks". The buttons invoke scripts which change what portion of the spreadsheet is displayed (i.e., moves to the next "screen"), change the screen scaling to make the display fit, write spreadsheet formulas and data, or call external code. The buttons are where most of the "action" occurs and where most of the calculation is done. The model is structurally dependent on scripting and the use of Fortran externals. About 50 pages of scripts are used and over 130K of compiled Fortran external code is used.

Because this database is intended for preliminary mission and vehicle trade studies, it provides both a means of obtaining detailed single point designs and a means of rapidly producing sets of parametric data and graphing that data.

Figure 20 shows the result of double-clicking the file "Parametric Database". Pressing the continue button takes the user to Figure 21 which is the main navigation screen. The Return button, which is present on all screens, always returns to the previous screen.

An example will illustrate how the various models are used. Pressing the Cryogenic button brings up Figure 22 and pressing the Large LOX/H_2 button brings up Figure 23. Since the LOX/H_2 model is implemented as external Fortran code, there are no equations under the numbers in the cells as would be expected in a spreadsheet. (For the models which are implemented as spreadsheet formulas under the cells there is no "Calculate" button.) The Calculate button in the upper left side of the screen must be pressed to produce numbers for the weights, lengths and performance. The independent variables, and the ranges through which each can be varied and remain within the validity of the model, are shown in the upper part of the screen on the yellow background. To examine a new case, change any or all of these independent variables and then press the calculate button. New values for the results will appear in the cells.

Pressing the "English Units" button changes the button name to "Metric Units" and changes the results (only) to metric units. Pressing the button a second time reverses the process. The Print (Report) button sets up for printing the page (without buttons) in portrait mode and stripped of color. The Print (Briefing) button sets up for printing the page in landscape mode and stripped of color. These buttons work the same on other screens. The page setup dialog box will always come up because Resolve script does not have a means to specify landscape versus portrait mode, so the user must click the appropriate icon.

The model can be used to generate parametric data and produce a table and selected graphs of that data. To do so, press the Graphs button and the parametric generation screen of Figure 24 will appear. This screen shows the variables which can be used for parametrics as titles within yellow buttons. The parametrics possible are one dimensional, only one variable can be varied at a time. To make a parametric run using one of the independent variables that are shown on the yellow buttons, choose a range of the variable to vary. Input its starting value and its ending value in the column "Variable to Change" (within the limits that are shown under each yellow button), along with the number of discrete points (11 maximum) to calculate (the variable values must be evenly spaced throughout the range which is why only the number of points, as opposed to the actual values, is input).

The column "Other Independent Variables" shows the values that will be used during the parametric run for the variables other than the one being varied. Use this column to change these values to those desired for the parametric run. These values start as the values from the previous screen, but they will change as parametrics are generated taking on the last value of the range used if they have been used in a previous parametric run. They should always be checked. When satisfied that the input is as desired, then press the yellow button that has the name of the variable that was chosen to vary. Pressing that button actually replaces the chosen independent variable in the screen of Figure 5, reads out the results, places them into a table and graphs, changes the variable again, reads out the results again, etc.

After the yellow button is pressed to generate the parametric run, a portion of Figure 25 appears. The table can be printed (Figure 26) and graphs can be individually accessed by pressing the yellow Weight, Lengths, and Performance buttons and then individually printed.

Propulsion System Database. The propulsion system database was developed using the Macintosh database FileMaker Pro, version 2.0v1 (published by Claris). It was developed on a Macintosh II fx running system 7 with the tuneup kit and using an Apple 13 inch color monitor.

The propulsion system database consists of two files: “Prop System DB” and “Prop System DB-Pictures”. They can be placed anywhere. The names of the two files must not be changed since the first is used as a look-up file by the second, and the second is referenced by name in scripts in the first. “Prop System DB” is the main file which contains all the data except two picture fields for each record. The two picture fields were separated because they are often scanned images using significant amounts of memory, and also by having two files, even when many more propulsion systems are included in the database, the FileMaker limit of 32 Meg per individual file should be avoidable.

The engine systems currently included in the propulsion system database are:

Space Transportation Main Engine (STME)
F-1
F-1A
J-2
J-2S
SSME
RD-170
Integrated Modular Engine (IME)
Space Shuttle Redesigned Solid Rocket Motor (RSRM)

To run the propulsion system database double-click on the file "Prop System DB". The opening screen of Figure 27 will appear. Press Continue and Figure 28 will appear. Pressing on any button will find all propulsion systems of the type represented by the button. For example, pressing "Cryogenic" will find only the cryogenic engines, pressing "Chemical" will find the cryogenics plus the solids, plus the hybrids, etc. Pressing "Propulsion Systems" will find all the records in the database. If the user presses a button for which there are no records of that type, a dialog box will appear and if Continue or Cancel is pressed, all records will be found instead of the null set of zero records expected. This is a quirk of FileMaker Pro.

The code is broken into five general classes of propulsion systems based on needing different reports for each kind of propulsion system: Liquids, Solids, Hybrids, Nuclear, and Exotic. The layouts for Liquids must be different from those for Solids since many parameters of one have no meaning for the other (e.g., mixture ratio, grain design). This structure is transparent to the user if the buttons supplied on every screen for navigation are used. In other words, when a liquid engine is selected and the Data Entry button is pressed, the user will go to the liquid data entry screen, not the ones available for solids, hybrids, etc. (which are different). Nonetheless, the actual internal structure is fairly complex and extensive because of the need for different report and entry formats. There are 160 layouts and 71 scripts used.

The result of pressing "Propulsion Systems" in the Main Menu (Figure 28) is shown in Figure 29 which is also the list of all currently available propulsion systems. An example of using the code is to select one of the propulsion systems from the figure (i.e., click on the engine name) and then press one of the five buttons across the top of the screen. The Print button simply prints the page (and works the same on all other layouts where it is present), the More Data button shows two additional lines of information for each propulsion system (thrust, specific impulse, weight, length, width, etc.) and is intended as a short technical summary of the systems in the database. The button with the "org chart" icon returns to the Main Menu (Figure 28). The Data Entry button goes to a set of layouts specifically designed to make data entry easy by gathering all the fields of data for one system in one place and eliminating any that are calculated from other data.

The Reports button goes to a screen like Figure 30. This screen shows the individual reports (layouts) available for each propulsion system. The reports are arranged into two sets – each containing the same information, but with some differences in arrangement – with one set structured for portrait mode presentation and called "Reports", and the other structured for landscape mode presentation and called "Briefing Charts".

Typical use of the code would be to go to the Main Menu screen (Figure 28), press "Propulsion Systems", choose an engine from the resulting Summary screen (Figure 29), press the Reports button and then use Figure 30 to look at the data (and print any of interest) by pressing individual reports. For example, pressing "Engine Performance 1" brings up the layout in Figure 31 (for a STME as an example). From this (or any other) report the user can print the report, return to the Reports screen, or return to the Main Menu.

After examining the various reports, the user might return to the Summary screen (Figure 29) and select another propulsion system and then look at its reports, and so on.

Figure 32 presents the reports for one propulsion system (the STME) as an example of the data available.

Documentation. Each of the two databases are available for use now. They are kept, in their most updated form, on a file server used to transfer data between MSFC and Rocketdyne. Bob Nixon of MSFC/PD can show anyone how to acquire them.

A Task Final Report (for the first year's efforts) has been separately submitted to MSFC/PD and is available from them. Each of the two propulsion databases, parametric propulsion database and propulsion system database, are described there. The descriptions include a user's guide to each code, write-ups for models used, and sample output. An appendix includes technical notes describing how to attach external code written in Fortran to both Resolve and to Excel. These procedures were developed during this year's effort with the Excel work done on Rocketdyne resources and the Resolve work done on a combination of contract and Rocketdyne resources. Interactions with tech support at Claris (the publisher of Resolve), Microsoft (the publisher of Excel), and at the publisher of the Macintosh Fortran compiler used, indicate that the use of Fortran externals with either Resolve or Excel breaks new ground. This capability will be extremely useful for the parametric propulsion database throughout the rest of this effort and should be very useful in general to anyone within the aerospace community using Macintosh computers.

SSME Upper Stage Use

The main objective of this study was to determine if the SSME can be used in an upper stage application in which an altitude burn for earth orbital insertion and an orbital translunar injection burn may be required. The SSME currently operates and performs cut off in a space environment; however, it starts at sea level in an ambient atmosphere. Also, the current tank pressures are higher than would be desirable for an upperstage. The key goals of this study were to determine viable methods for starting the SSME in an altitude environment and restarting it in an orbital environment with minimum changes in utilization of the engine system or hardware.

A common start sequence for both altitude and orbital conditions was a key objective of the study. By maintaining a common start sequence development costs can be minimized.

The impacts on the engine start differ for the altitude start versus the orbital restart cases and are summarized in Figure 33. For the altitude start case, the thermal conditions are the same as the current ground start. However, the pressures are quite different. The gravity head is absent and both the fuel and oxidizer inlet pressures are reduced from the ground start case. More importantly the pressure reduction is not in the same ratio for both the fuel and oxidizer, which strongly affects underlying control assumptions of valve proportionality.

For the orbital restart case, the pressure conditions are different from the ground start case but they are similar to the altitude start case. But now it is the thermal conditions which are very different from the ground start case. The engine has been fired and shut down and the engine is in orbit which changes the thermal conditions from the ground start case. The change is not the same, in direction or amount, for all components. Different components are changed in different ways. Some are hotter than the ground case, some are colder. Consequently, the mixture ratio assumptions used in the control scheme are affected.

There is one additional impact for the restart case. The start environment of the engine has been changed from the ground start case because the engine has been fired and has been shut down in a vacuum. Water can be formed during the shut down and potentially form ice and change the start-up characteristics of the turbines. However, the combustion quenches after valve closure but before the purges during shut down and water formation is only possible during combustion. Additionally, vacuum means there is zero back pressure on the system and that the water vapor pressure is always above the pressure in the chamber and turbines. Consequently, water and ice formed may evaporate and/or sublime.

Altitude Start. The input conditions used for the altitude start case were based primarily on the Apollo program Saturn V vehicle, S-IVB stage propellant inlet conditions and to a limited extent on engineering estimates of minimum pressures which would be viable for the operation of the SSME engine. The oxidizer inlet pressure was reduced from the shuttle external tank value of 107 psia (which is about 80 percent gravity head) to 40 psia, which was the value used on the S-IVB. The fuel pressure was reduced from 45 psia down to 32 psia which corresponds to a shift providing approximately the same delta pressure across the system (45 - 14.7 versus 32 - 0). In addition, 32 psia represented a lower bound with respect to fuel pump performance in that a margin must be maintained for NPSH above the vaporization pressure at the pump inlet to provide for engine and start variables.

The intent of the study was to develop a start sequence which minimized adverse changes to the engine hardware or operation as well as minimized changes which would void the 500,000+ seconds of SSME experience base.

Engine operation for the altitude start would occur within five minutes of launch; therefore, changes in hardware conditions which influence the start would be relatively small. The critical condition primarily being hardware temperature being near ambient. Confidence in this assumption was high since the engine hardware mass is greater than 7000 lbs which has soaked heat prior to launch and which was protected during the boost phase (aft skirt enclosure). The purges prior to altitude start were not foreseen as a significant driver to stage design since provisions for purges would already be required for safety purposes to avoid propellant accumulation in interstage connection compartments.

Safely establishing and maintaining proper mixture ratios for the two preburners and the main combustion chamber (MCC) to ignite and sustain combustion is a key objective during the ground start phase of the current SSME. In lowering the inlet pressures of the propellants, the mass flow rates through the engine system were altered and hence changed the mixture ratios at any given time up until priming of the oxidizer preburner. To achieve an altitude start sequence as close as possible to the current SSME, the valve schedule were adjusted. Three key oxidizer valves, along with the main fuel valve, control the start sequence. Changes were made in valve sequencing for these valves for the altitude start case.

Additionally, the preburner and main combustion chamber igniter systems are supplied propellants by separate lines and would require reorificing to account for the lower inlet pressures.

The SSME transient start model was run with these changes and several iterations were made to evaluate required changes and resulting characteristics of the start phase for altitude start. The results are shown in Figure 34 which shows the main chamber pressure versus time during the start sequence. The analysis determined that only minor valve schedule changes were required (and only in the first 2.2 seconds of the 4.2 second start phase), as shown in Figure 35, to accommodate the lower propellant tank pressures with the total time to reach mainstage being unchanged. The priming sequence occurred in the same order with only slight timing delays from the current start priming sequence. The high pressure fuel turbopump turbine temperature spike was reduced to provide greater margin for the turbine blades in the event of start variances. The high pressure oxidizer turbopump turbine temperature spike, which enhances propellant ignition with the lower turbine gas mixture-ratio/temperature, was raised slightly to improve ignition margin.

The primary difference between the operating conditions for the current start versus the altitude start was a lower set of pump inlet pressures. The analysis performed showed that an altitude start is feasible with the SSME engine with only modifications to valve sequencing (both as to position and timing) and some reorificing, at least with inlet pressures of 32 psia on the fuel side and 40 psia on the oxidizer side.

The net effect of these lowered pressures was to delay the initial bootstrap rate, although the overall time to bootstrap the engine and the time to full mainstage remained the same as the current start (see Figure 34).

Orbital Coast Thermal Analysis. Thermal modeling of the SSME during its orbital coast was conducted to acquire the needed inputs for the orbital restart analysis. A total of 24 cases were run to characterize and define sensitivities of the SSME components critical during the start phase. The fundamental assumptions consistent for all modeling cases of the engine were that the engine/stage would rotate along the X-axis and that the inertial path angle would remain zero degrees during the coast phase. The coast phase orbit for the modeling was approximately 95 nautical miles and circular.

A baseline case was established in which the engine temperatures were determined over a period of 4.5 hours after engine cutoff. No recirculation or modifications to the engine system were included in this case. The need for propellant recirculation was identified and the trends of the component temperatures were established. Temperatures of components critical to ignition were found to typically approach ambient after one hour in orbit. The preburners lag behind, but the turbines remain above 300 F even after 4.5 hours. The liquid hydrogen and LOX turbopumps were found to heat up significantly above the required propellant temperatures and pressures for ignition.

Figure 36 summarizes the main results of the component thermal analysis. The thermal analysis of the engine components revealed that significant variations and responses occurred once the engine shut down from its first burn. Components with large surface areas and low relative masses, such as the nozzle, responded with dynamic behavior, while components buried down in the middle of the engine such as the fuel preburner injector and oxidizer preburner oxidizer supply line, tended to be less dynamic. Most components trended toward ambient temperature. However, two component areas critical for controlling mixture ratio in the initial phase of the start sequence were found to lag behind the remainder of the engine components.

For example, the nozzle, as shown in Figure 36, represents components which experience large variations in temperature as the vehicle goes from the solar side to the shadow side of the earth. Consequently, cases were picked to specifically examine the impact of these variations on the restart (the two dashed lines at 7,500 seconds and at 10,100 seconds). Both extremes had sufficient energy for the start and neither extreme hampered the start sequence.

There is also a class of components, exemplified by the FPB injector in Figure 36, which start too cold to allow a start but which steadily rise in temperature. Components of this type would require heating early, but at some temperature (i.e., at some time after the orbital insertion burn) would be warm enough to allow restart without that particular component needing additional heating. 7,500 seconds was found to be the time necessary for components of this type (see the example of the FPB injector in Figure 36 and note that the temperature does not necessarily have to rise all the way to the ground start conditions for all components).

The third class of components, also shown in Figure 36 as the lowest two curves, are those that started cold and stayed cold. If these components are nominally at ambient ground conditions for the SSME ground start, then they will need heating, though not necessarily to full ground ambient, to keep the restart within the SSME experience base. This is particularly true of those components which affect the mixture ratio during the earliest part of the start sequence.

Additional cases were run to define and parametrically study the recirculation flowrates required to keep the turbopumps at the required temperatures and pressures necessary for ignition. A flowrate of 1.0 lbm per second is recommended for both the LH_2 and LOX systems. To provide the earliest possible restart of the SSME the recirculation should be initiated immediately after boost phase cutoff.

A temperature fluctuation was found to occur on the LOX pumping system corresponding to the solar cycling. During the solar portion of the orbit the temperature rise reduced the Net Positive Suction Head (NPSH) margin significantly for the LOX turbopumps. A study of the effects of insulation and thermal control paint was conducted to improve the duct wall temperatures at the pump inlets and thus the required NPSH margins. Figure 37 summarizes the effects of insulation and thermal control paint on the temperatures just upstream of the LOX and H_2 turbomachinery (and thus on the tank pressures needed for restart). Recirculation flow is assumed at one lbm/sec. If no insulation is used on the LOX side (insulation is already present on the H_2 side on the nominal SSME) and no thermal control paint is used, then the baseline conditions of 181.4 °R on the LOX side and 40.3 °R on the H_2 side are achieved at the worst time of the solar portion of the orbit. These temperatures are too high if a low stage pressure is desired.

The second case shows the effect of adding insulation on the LOX side but it produced little improvement. The model used does not take advantage of the insulation's nickel coating as a reflector. Although not all of the energy input is in the wavelength band where nickel is reflective (or, for that matter, where the paint is reflective), most of the energy is in such a band. Consequently, the third case shows the best that could be achieved if the nickel coating is very clean and smooth. Reality would lie somewhere in the band between cases two and three. Case four analyzed the effect of a thermal control paint, without insulation on the LOX side, and not using any additional reflectivity from the nickel (it would be painted over). The paint is very effective at reflecting energy from sunlight and reflected sunlight (Earth albedo). It was assumed totally ineffective against the Earth's blackbody radiation. This case of using paint produced temperatures which allow low tank pressures throughout the orbit.

The recirculation flow could be used to do the same job as the paint by carrying away the heat (the paint prevents the heat from arriving). The last case shows the flowrate needed to achieve the same effect as the paint.

Further thermal modeling was conducted to determine the propellant recirculation requirements for longer duration coast periods. Recirculation was started at 1, 2, and 3 hours after boost phase cutoff to parametrically evaluate subsequent chilling times. The results indicated that a chilling period of approximately 1.5 hours of recirculation at 1 lbm per second would be required to achieve conditions required for restart.

Based on the results of the thermal analysis the moisture which may be present in the engine at boost phase cutoff will most likely be removed from the system since the temperatures of the components warm-up to near ambient temperatures.

Orbital Restart. A restart simulation analysis, utilizing the SSME transient model, was conducted. A common start sequence for the altitude start and orbital restart was maintained for all cases analyzed using the sequence already determined for the altitude start case. In general, the engine behavior became closer to nominal with longer coast time periods before restart. The results suggested that restarting the engine at coast periods greater than 7,500 seconds (125 minutes) was a reasonable option since only about 0.05 second of delay was experienced with respect to the altitude start case and that limited thermal conditioning of engine components was required. For comparison, Apollos 8, 10, 11, 12, and 13 all restarted between 8,500 and 9,500 seconds after orbital insertion, and Apollo 9 restarted after 16,500 seconds.

Figure 38 shows the results, in terms of chamber pressure versus time, for the orbital restart cases at both 7,500 and 10,100 seconds after the end of the orbital insertion burn.

Thermal conditioning was necessary. Both recirculation of propellants through the pumping elements of the engine system and heating two key areas of the preburner propellant feed system were required. The heating was required to provide a more robust start capability for the engine at any time after 7,500 seconds of coast period. Minimum and maximum nozzle temperature fluctuations predicted during the coast period, after 7,500 seconds, were evaluated and found to have no significant influence on the restart characteristics of the engine.

Component thermal sensitivities were explored on a limited basis typically for those components requiring warmer temperatures to achieve a reasonable restart time.

Some components were predicted to be very close to Earth ambient by 7,500 seconds, and thus were not examined for thermal sensitivities. Additional work is recommended to further evaluate all components for thermal sensitivity for engine restart.

Engine restart prior to 7,500 seconds was evaluated. If earlier restart capability is required, direct heating of four additional components (oxidizer preburner injector, fuel preburner injector, fuel preburner fuel supply duct, and oxidizer preburner fuel supply duct) would be needed.

Figure 39 summarizes the study conclusions for the altitude start and the orbital restart cases.

Inlet Pressures. The inlet conditions for the liquid hydrogen must satisfy specific pressure and temperature requirements in order to provide sufficient net positive suction pressure (NPSP) for the low pressure fuel pump and the high pressure fuel pump. Figure 40 summarizes the start conditions for all the cases examined in the study. Minimum NPSP curves are plotted for both 109% and 100% power levels along with the vapor pressure curve for liquid hydrogen. The operating point for the engine must be above the NPSP curves to prevent detrimental cavitation from occurring in the pumps. The shaded box marked "SSME Ground Start" shows the specification start conditions for the current SSME start at liftoff. The altitude start case that was evaluated for a tank pressure of 32 psia is shown. The altitude case represents the lowest pressures that could be used with an SSME assuming that the hydrogen is delivered at the highest temperature of the current ground start conditions. The restart cases are shown, all for the worst time during the orbit which is on the sun side. The case without paint and without assuming any heat reflection from the nickel coating of the insulation is marginal. The restart case using thermal control paint produces significant margin.

The oxidizer system requirements for pump inlet conditions are similar in character to those of the fuel system and are shown in Figure 41. There is also a remote possibility that helium ingestion can take place in the oxidizer system, so that NPSP requirements for that condition are also included on the chart. The ground start is above 100 psi due to the head contribution provided by the LOX tank location in the top portion of the Shuttle external tank. Not all of the pressure is needed to satisfactorily operate the low pressure oxidized pump. The altitude case evaluated was basically the ground start case without the gravity head. The altitude case is shown and falls below the 109% power level NPSP line but above that needed for a start to 100%; however, starting above 100% power level was not required for the upper stage applications that was defined. If the need arose, a two step start could be implemented to allow acceleration head to be established prior to throttling above 100%. The restart cases for a restart on the sun side of the orbit are also shown. The basic case, without insulation or paint, is inadequate unless the pressure is raised significantly to about 47 psia. (Restart in the shadow side is possible at 40 psia.) However, use of thermal control paint produces significant margin for restart at 40 psia thus allowing reasonable sun side restarts.

All the cases shown in Figures 40 and 41 assumed the use of recirculation flows of 1 lbm/sec for both the fuel and the oxidizer.

Development Plans. The program needed to develop and certify the SSME for upperstage application can be accomplished with low risk and relatively low cost compared to a new engine program. Key testing can be accomplished in a minimal cost demonstration program to provide an early understanding of the risk involved before development and certification of SSMEs for upperstage use is started.

The ground rules and assumptions which were used for estimating the program costs were: all costs are in Fiscal Year 1992 dollars; the cost of production engines for the new vehicle is not included; the demonstration program and development program are conducted in series and transition immediately from one to another; engine unit costs are based on a total production rate of six per year; only minor changes, such as reorificing of igniter propellant feedlines, adding insulation/thermal control paint, reducing insulation on the nozzle, and incorporating a LOX propellant recirculation system are required; procedural changes for the engine are assumed to be required as well; the engine used for the demonstration is upgraded and used as the first development engine. Propellant costs are not included in the cost estimate as they are typically furnished by the customer. The total program cost of $174.8 million does not include fee. The schedule assumes that one test stand at the NASA Stennis Space Center is available and that 130 tests are needed between the Arnold Engineering Development Center and SSC. Assuming production of flight engines occurs 2 1/2 years after the program is initiated, initial launch capability is viable in 5 1/2 years from program start.

Documentation. Results of the altitude start case evaluation were included in the first review briefing book (17 June 1992), results of the orbital thermal analysis were included in the second review briefing book (1-2 October 1992), and an overview of the entire task, with emphasis on the restart analysis, was included in the final program review briefing book (17 March 1993). A detailed Task Final Report has also been submitted as a separate document. All of these are available from MSFC/PD.

CERs for Liquid Propellant Rocket Engines

The objective of the CERs for Liquid Propellant Rocket Engines was to provide NASA/MSFC with parametric cost estimating relationships (CERs) at the engine and major subassembly level for estimating development and production costs of chemical propulsion liquid propellant engines in the vacuum thrust range of 20 klbs to 2,000 klbs.

The task output will be useful to parametrically estimate the development and production cost of (1) new liquid propellant rocket engine, (2) check the validity of rocket engine costs provided by contractors, and (3) identify those technical parameters which are rocket engine cost drivers.

The cost modeling approach was divided into two parts: (1) production and development cost models for engine systems and (2) production cost models for major engine subsystems such as combustion devices and turbomachinery. All models contained parametric Cost Estimation Relationships (CERs) which gave cost as a function of size and complexity attributes. Cost Breakdown Structures defined the individual cost elements to which the CERs are applicable.

The cost models are to be understood as engineering models and were not based on regression analysis, since they were using only a few data points. The CERs were anchored (calibrated) with the technical and cost data of Rocketdyne's engines. Cost data were obtained from company records, not from government sources, for traceability and "purity" reasons.

Production rate and quantity effects were also obtained from Rocketdyne's historical database. The influences of these significant factors on hands-on labor, support labor and material cost were combined using Rocketdyne's process-oriented production cost model.

The Development Cost Model uses "real world" cost drivers such as Engine Complexity, Maturity, Test Frequency, Process improvement factors, etc. Considerable insight into development cost driving parameters were obtained by analyzing in detail three available engine development program cost breakdowns.

The cost models were intentionally simple in order to be useful in the early program phases of future engines when few parameters are known. Technical and programmatic descriptions of an engine or a major subassembly are input into the cost model and are translated into cost parameters via cost models which are comprised of CERs and engineering estimates. The cost models quantitatively characterize the engineering and manufacturing knowledge.

The cost database for the models contained available historical cost and technical information of six actually produced engines and other relevant data sources. For example, the detailed cost analysis of the Advanced Space Engine for OTV applications (staged combustion cycle) was used in the construction of the size-dependent engine production cost parametrics and in the engine development cost model. Peacekeeper Stage IV data was used in the determination of rate and quantity cost improvement curves; Lance engine data was used for quantity cost improvement curves.

The generic cost breakdown structure used the LCC cost elements of a rocket engine as shown in Figure 43. Of the five cost categories (DDT&E, Production, Preplanned Program Improvement, Operations and Support, and Disposal), only the first two were defined by the cost models of this task. The development cost element encompassed mainly engineering, engine and component hardware, acceptance testing and program management. The certification and reliability demonstration cost elements contained engineering, testing, hardware and propellant costs.

The production cost category included all hands-on and support manufacturing labor, procured hardware from subcontractors and raw material, engineering support, production management, acceptance testing, test propellants, and government support.

The elements contained all engine contractor and component subcontractor cost items through general and administrative expenses (G&A), but excluded engine contractor fee (subcontractor fee is included.).

Production Model. The production cost model shown in Figure 44 was a deliberately simple model. Primary inputs are vacuum thrust, thermodynamic engine cycle and type of propellants. A basic CER has been constructed which relates TFU cost at 30 units per year (in 1992$) to these three parameters. Eight adjustment factors were generated to modify this TFU cost for chamber pressure (Pc), reusability (REUSE), manufacturing improvement (IMP), production rate (RATE), production quantity (Q), automation effect (CIM), dollar escalation (ESC), and contractor fee (FEE). All adjustment factors are multiplied with each other and with the TFU cost from the CER to yield the unit production cost under the input conditions.

The adjustment factors are:

The chamber pressure adjustment factor is of parabolic nature and adjusts the TFU cost between pressures of 500 and 3000 psi. It is a second order magnitude effect on cost.

The reusability factor equals one for long life LOX/LH2 staged combustion engines with SSME-similar life characteristics. It is less than one for expendable LOX/LH2 staged combustion engines. This factor is greater than one for reusable gas generator engines, since the CER was based on expendable gas generators. "Expendable" engines are those designed for less than twenty missions (i.e., it includes short-life reusability).

The producibility (manufacturing) improvement factor is one for all historical engines; it is less than one for the upcoming new generation of "low cost" engines.

The production rate factor equals one for thirty units per year; it is greater than one for lower rates. A cost improvement curve approach is used with rate substituting the normally used quantity.

The production quantity factor equals one for TFU cost, and is less than one for higher quantities. A normal cost improvement curve factor is used (also called a "learning curve").

The automation effect considers Computer Integrated Manufacturing (CIM) with a high degree of automation for production rates of 50 or more units per year. Equal cost sharing of the facility investment between industry and government is assumed.

The escalation factor is one for FY 1992 dollars, it is larger than one for prior year dollars, and lower than one for FY1993 and future year dollars.

The fee factor is one if cost is desired as cost model output, or greater than one (e.g., 1.10) if price is the desired output

All factors are independent of each other.

The net effect is that a top level parametric cost model was generated for pump-fed liquid propellant rocket engines which allows the unit cost prediction of rocket engines with few known parameters.

The model is based on engineering analyses of historical data. It is valid in the thrust range of 20 Klbs to 2000 Klbs; i.e., mainly for booster engines and high thrust upper stage engines. It should not be used below 20 Klbs thrust. The other key parameters besides thrust are thermodynamic engine cycle and type of fuel. The estimated uncertainty of the model is ± 30%; this is the generally accepted uncertainty band of parametric cost models with relatively few inputs.

The effects of cost improvement for production rate and quantity (cost improvement curves) were incorporated, based on historical data. The combined effect for rate and quantity has significant influence on unit cost.

A producibility improvement factor was established, based on several detailed engine design, manufacturing and cost analyses performed at Rocketdyne during the last three years. For expendable engines, producibility improvements have a significant effect (factor of 2 to 3) on production cost.

Development Model. Figure 45 is an overview of all rocket engine development phases. The engine development cost model covers all phase C/D contractor efforts, from the end of phase B to the flight phase. Phase C/D starts with the fully defined requirements for the engine and ends with successful completion of the single engine certification program. After phase C/D the engine is certified for first flight.

The build-up of all cost elements which make up the total engine development cost to the taxpayer are shown in Figure 46. (The cost elements are presented in a nested box format, as suggested in the "Systems Engineering Management Guide," Defense Systems Management College, 1986 edition, rather than the more conventional WBS-type tree structure.)

The innermost nested box represents the items covered by the engine development cost model, it excludes only contractor facility costs. Main propulsion test article (MPTA), flight engines and initial spares hardware are development items which are excluded from the cost model. Also excluded are government costs, propellant costs and contingencies. Government support in the past has been in the order of 15% and propellant costs for development have been about $200M (1993) each for the F-1, J-2 and SSME engine programs.

An analysis of the development program cost distribution of two engines (F-1 and J-2) has disclosed that the majority of the cost, more than 70%, is due to failure mode elimination. This accounts for the iterative test, analyze, and fix (TAAF) cycle of the component and engine development program. Only 2% was expended for the initial design effort, 15% for engineering design and analyses, mainly in the early part of the program, and 10% for qualification, reliability demonstration and certification.

This indicates that a representative development cost model should address the number of tests required for the TAAF cycle as a key parameter. It also indicates that the number of tests is a cost driver which must be reduced to result in lower development costs. A development cost model which is keyed to engine size would not lead to appropriate CERs.

The core of the cost model consists of the parameter "number of tests required." This parameter directly determines the test labor cost and the required quantity of development engines. Together with the engine unit cost obtained from the production cost model, the number of development engines defines the total development hardware cost. The cost of (1) design engineering/ analysis and (2) tooling, ground support equipment and special test equipment needs to be added to hardware and test cost to sum up to the development cost. Program management cost and fee are usually estimated as a percentage of the development cost. The cost elements are aggregated to total development cost as indicated by the innermost nested box delineated in Figure 46.

The development cost model is mainly based on three engines: F-1, J-2 and SSME. Only for these three engines, a somewhat detailed breakdown of development costs was available.

The cost model is applicable to liquid bipropellant, pump-fed rocket engines in the 20 to 2000 Klbs thrust class.

There are eleven parameters that define the engine development cost model inputs. They consist of (1) seven adjectively determined engine complexity and maturity indices and process improvement and tooling availability factors, and (2) four objectively determined programmatic and unit cost inputs.

The first group of seven adjective factors are judgmental in nature, but with a graduated scale given for metrification in a series of charts in the Task Final Report which is available as a more comprehensive documentation. The second group of four parameters are objective quantitative inputs.

The factors are:

Adjective Factors

- Engine Cycle/Internal Environment Complexity (CYPLX)
  - Measure of Cycle Complexity
- Engine Design/Manufacturing Maturity (ECMPLX)
  - Measure of Engine Maturity
  - Measure of Technology State-of-the-Art
- Tooling Availability Factor (TAVAIL)
  - Measure of Retooling Degree
- Test Quantity Process Improvement Factor (PIF1)
  - Expression of Testing Philosophy
  - Measure of Certification Approach
- Test Process Improvement Factor (PIF2)
  - Measure of Testing/Setpup/Post Test Simplification
- Design Process Improvement Factor (PIF3)
  - Measure of Design Automation
  - Degree of Availability/Use of Advanced Design Technology
  - Degree of TQM Implementation
- Tooling Improvement Factor (TIF)
  - Measure of Tooling Modernization

Objective Factors

- Test Frequency in Tests/Month (TFRO)
- Development Engine Fabrication Time Span (DET)
- Theoretical First Production Unit Cost (TFU)
- Anticipated Engine Production Rate (R2)

The actual algorithms used by the model are shown in Figure 47.

Summary. Top level parametric production and development cost models were generated for pump-fed liquid bipropellant booster and upper stage rocket engines in the 20 Klbs to 2000 Klbs thrust class.

The models cover production and full scale development costs and are based on thorough engineering analyses, not regression analysis, of data from historical

Rocketdyne engines, potential engine derivatives and proposed new engine concepts. The models are not weight-based, but depend on thermodynamic cycle, propellant type, engine complexity, engine maturity and other design parameters. The models are simple, with a documented transparent rationale, and the CERs were incorporated into a spreadsheet. The estimated uncertainty of both the production and development cost models is ±30% within the stated CER limits and within the thrust range of 20 Klbs to 2000 Klbs.

Programmatic factors for production rate and quantity, and for development test frequency and hardware fabrication rate are included.

The models make use of adjective and objective parameters. For the adjective inputs, metric scales are given to convert them into numerical values. The adjective inputs require good engineering understanding of rocket engine design and manufacturing principles.

Several process improvement factors are incorporated to make the historical data based cost models applicable to new reusable advanced performance and/or to low cost engine concepts.

The validity and reasonableness of the cost models was successfully checked against STME data and against current manufacturing and programmatic analysis results of new engines.

The cost models are extendible to cover tripropellant engines or engines with other propellants if moderate additional effort is expended to parametrically model rocket engines at lower tiers (i.e., at subsystem and major component levels).

Documentation. A separate Task Final Report has been submitted and is available from Steve Creech of MSFC/PP03.

Advanced Low-Cost Engines

The objective of this task is to produce concepts for future engines which are both high in performance and low in cost (development, production, and operation) in order to enhance vehicle performance. Advanced means both high specific impulse (at least as high as SSME, preferably higher), produced primarily through high chamber pressure, and low weight to produce higher thrust-to-engine-weight than current engines. The engine concepts are being optimized for the Single-Stage-to-Orbit (SSTO) application with the vehicle as defined by NASA-LaRC for Option 3 of the Access to Space Study.

Concept Definition. The study defined six engine cycles for study. Then baseline component parameters (such as turbine material and operating temperature) were chosen along with sets of variations on these parameters. A single position bell nozzle was chosen as representative (i.e., a different choice would not have produced differentiation among the cycles and concepts although it would have changed the overall vehicle performance). A fixed exit pressure of 4 psi was chosen since previous studies had shown that value to produce near optimum vehicle performance for a single position bell nozzle. Consequently, engine area ratio was a set function of chamber pressure as shown in Figure 48. Figure 49 shows the baseline component parameters, their ranges examined, and other specific technologies included in the baseline.

The six engine cycles examined included one representative open cycle (a gas generator cycle, Figure 50) and five closed cycles. The open cycle had the lowest engine weight, but also had a significant performance penalty in comparison to the closed cycles.

In a closed cycle the amount of energy which can be extracted to pump the propellants, and thus increase chamber pressure and engine specific impulse, is dependent on the regenerative heat from cooling, how much of each propellant is available to the turbine, and whether chemical energy (i.e., preburners) are used to increase the energy of the turbine flows. The five closed cycles explored this range of energy extraction capability.

The first closed cycle (Figure 51) was a full flow mixed preburner cycle using individual preburners to power the fuel pump and the LOX pump. The fuel preburner was fuel rich and the LOX preburner was LOX rich. Thus potentially all of both the fuel and LOX flows were available. This cycle could extract the most energy for pumping and thus was capable of the highest chamber pressure. Because it had the most and the largest powerhead components it was also the heaviest cycle of the five at a given chamber pressure and nozzle area ratio.

The next cycle, both in the ability to extract energy and in weight (i.e., second heaviest), was a cycle which used all of only one flow (in this case, H_2) but also used preburners for both the fuel and LOX pumps. This staged combustion cycle (SSC) was very similar to that used for the Space Shuttle Main Engine (SSME). Figure 52 shows this cycle.

The third closed cycle was one which also used all of one flow (H_2), but only one preburner. The preburner was used to power the fuel pump because the fuel pump needs more horsepower than the LOX pump, and, consequently the cycle could extract more energy if the one preburner was used on the fuel side. A fuel expander (fuel using only the energy from regenerative cooling) was used to power the LOX pump. This cycle, the hybrid cycle, is shown in Figure 53.

The inverse of the hybrid cycle was also examined: a preburner powering the LOX pump and a fuel expander powering the fuel pump. This cycle is illustrated in Figure 54.

The last closed cycle examined was one using fuel expanders to power both the LOX and fuel pumps. This cycle had the least ability to extract energy and thus had a lower maximum chamber pressure. However, it also had the lightest engine weight of the closed cycles at a given chamber pressure and area ratio. The expander cycle is shown in Figure 55.

The defined cycles were examined from a chamber pressure of 1000 psi to the limit the cycle could produce by using the Rocketdyne balance code. At each chamber pressure the pump and turbine stages were varied and both pump discharge pressures and engine weight were minimized.

Engine Weight Calculations. Engine weights were calculated for all six cycles as a function of chamber pressure. The weights included all the engine systems that would be in a reusable engine such as the SSME. Thus controllers, line insulation, gimbal attachments, drain lines, etc. were included. Installation specific systems such as the gimbal actuators and the engine heat shield were not included in the calculated engine weight. However, these items were explicitly calculated by the LaRC vehicle code. Figure 56 shows the methodology used to calculate the engine weights. They were calculated for two levels of technology: one with minimal advancement over that used in the SSME (referred to as the "bracketing" weight set since it should be an upper bound on a new engine), and one with a moderate number of near and midterm technologies included in the new engine (referred to as the "aggressive" weight set).

The new technology used was jet pumps as the boost pumps, turbomachinery specifically designed to lower cost and weight, EMA valves, and a limited use of advanced materials for the thrust cone, gimbal bearing, H_2 valve bodies, H_2 pump, gimbal actuator attach bracket, support struts, and the nozzle jacket. Advanced materials were used for few major engine components and thus there is probably weight margin in the estimate compared to methods which emphasize material approaches to lowering engine weight.

Figures 57 and 58 show the engine weights for both sets of technology assumptions. The weights shown are for the highest turbine temperatures (which use Si_3N_4 turbine wheels and represent a new technology). Figures 59 through 61 shows the weights for the FFSCC, SCC, and hybrid cycles with varying turbine temperatures from 1,000 to 2,500 °R.
As described later, the weights were validated through a bottoms up CAD design. The result was that the weights will be less than those shown for the "aggressive" set.

SSTO Performance. Twenty sets of resulting engine characteristics (weight, thrust, specific impulse, and mixture ratio) were sent to NASA LaRC to determine the vehicle gross and empty weights for a 25K payload to 220 n.mi. at 51.6° inclination. A non-linear regression analysis was performed on these results and the resulting equation used to predict the other engine cases.

Figure 62 shows the vehicle results for the six cycles. The results showed that chamber pressure was the most significant driver of vehicle performance (through its effect on area ratio and thus on vacuum specific impulse) - but only up to 4,000 psi. Above that value the increase in specific impulse was offset by the increase in engine weight and the vehicle empty weight was essentially constant.

The gas generator cycle produced vehicle empty weights about 13-15% higher than the closed cycles despite the gas generator's lower engine weights due to its much lower vacuum specific impulse.

Most of the closed cycles behaved the same with steadily decreasing vehicle weight until about 4,000 psi. The only exceptions were the dual expander and the reverse hybrid cycles which could only reach chamber pressures in the 1,500 to 2,000 psi range. The other three cycles could all reach the optimum chamber pressure range and showed only minor differences between cycles because the engine weight differences between cycles was small.

The effect of turbine operating temperature on the vehicle empty weight was also examined. Figures 63 through 65 show the results. The results were that all three cycles (FFSCC, SCC, and hybrid cycle) could operate at temperatures below those of the current SSME and thus did not benefit from the higher temperature capability of Si_3N_4 as a turbine blade material. At a chamber pressure of 4,000 psi, the FFSCC and SCC could operate in the 1,000 to 1,200 °R range for both the fuel and the LOX turbines, whereas the hybrid cycle could operate at about 1,600 - 1,700 °R fuel turbine operating temperature.

Based on these turbine temperature versus vehicle dry weight results a baseline engine configuration, as shown in Figure 66, was determined for each of the three cycles. These configurations minimized the turbine operating temperatures to improve reliability and life; to lower weights and complexity though the use, except possibly in the hybrid cycle, of uncooled powerhead components; and to allow margin for requirement changes or analysis errors.

Margin Study. A series of sensitivities was generated for thrust margin (+5%), throttling capability (5:1), changes in turbopump efficiencies (-5%), changes in thrust chamber pressure drop, and changes in general system pressure drops (+5% of pump discharge pressures). The use of all these sensitivities together is a severe test of the margin capability of the cycle.

Three cycles were examined: FFSCC, SCC, and hybrid. The baseline cases were for 4,000 psi chamber pressure for each cycle. Figures 67 and 68 show the vehicle dry weight and the fuel turbine operating temperatures for the three cycles and the different margins. The FFSCC could accommodate all of the sensitivities together with only an increase in turbine inlet temperature of 1,100 to 1,350 °R on the fuel side and 1,100 to 1,248 °R on the oxidizer side. This produced a 385 pound weight increase resulting in a 3.8% increase in vehicle empty weight. The SCC could also meet all the sensitivities together, but the turbine inlet temperature increased from 1,200 °R to 1,822 °R on the fuel side and 1,100 °R to 1,610 °R on the oxidizer side. These increases, unlike those for the FFSCC, would require changing to the use of cooled preburners, turbines, and hot gas ducts. The SCC engine weight increase was 788 pounds which resulted in an 8.0% vehicle empty weight increase.

The hybrid cycle could not easily meet the throttling sensitivity without lowering the chamber pressure below the nominal of 4,000 psi. Only by using silicon nitride as a turbine material and a 2,500 °R turbine inlet temperature was it possible to maintain the 4,000 psi chamber pressure. Using a pintle injector for the preburner also came close (3,961 psi), but with a higher engine weight. Both would require cooled powerhead hardware.

The net effect of the margin study was the choice of the FFSCC as the final baseline case for the weight validation and layout efforts.

Life Cycle Costs. The purpose of the life cycle cost (LCC) analysis was to determine the impact of LCC on the engine selection and, in particular, to determine if vehicle dry weight was an acceptable surrogate for total LCC. The engine life cycle costs were calculated using the cost models for engine development and production described earlier in this report and were generated for three points with one engine cycle (FFSCC) and for single points with each of two other cycles (SCC and hybrid). The engine operations cost estimate was calculated separately. MSFC then supplied the vehicle LCC. Figure 69 shows the relative engine LCC. Note that it minimizes around 3,000 psi and does not have the shape or trends of the vehicle dry weight results. Figure 70 shows the relative LCC for the entire SSTO. It shows the same trend as the vehicle dry weight results. However, the LLC is much less sensitive than the vehicle dry weight. For example, a 14.7% change in vehicle dry weight produced only a 2.4% change in total LCC.

The net result is that vehicle dry weight can be used as a surrogate for total LCC but always accounting for the much lower sensitivity of the total mission LCC.

Weight Validation. A design layout was generated using a FFSCC at 4,000 psi chamber pressure as a baseline. The layout was then used for a more detailed weight determination as well as producing configuration drawings. The detailed weight determination produced a weight of 4,413 lbm. The previous estimate for this configuration had been 5,003 lbm.

The weight estimate was a bottoms up CAD design of the entire engine including all major and minor (e.g., drain lines, heat shield attachment flange) components. A detailed weight statement of the SSME was used to that no element of a real, fielded, reusable engine was unaccounted for. Figure 71 shows the design point and characteristics of the engine. Figure 72 shows the procedures used for the weight calculation and Figure 73 shows the results (more detail is available in the Task Final

Report). This detailed weight determination is a much better estimate than that made earlier in the program. Figure 74 summarizes the resulting engine.

The net result of the detailed weight determination is that the vehicle dry weight results generated during the program remain usable although conservative, i.e., the vehicle dry weights will be less than had been shown. Figure 75 shows the improvements that will occur from the corrected weights. The effect due to the engine not needing a He purge (an effect which was not accounted for earlier in the vehicle results) is also shown in Figure 75 and is significant.

Tripropellant Engines. Within the Option 1 effort a single tripropellant engine concept is being used to examine vehicle performance relative to the bipropellant engines already defined. A bell annular configuration was chosen. This configuration, illustrated in Figure 76, uses a core LOX/H_2 engine and an annular ring using LOX/RP propellants attached just below the throat of the core engine. Only a fixed bell nozzle was examined. The total sea level thrust is 421,000 lbf (the same as the bipropellant engine designs) and the thrust split is 0.7 (1-(total mode 2 vacuum thrust/total mode 1 vacuum thrust)). Two pressures, 3,000 and 4,000 psi, were examined as was the use of both a FFSCC and a gas generator cycle for the LOX/RP side (the LOX/H_2 side used a FFSCC for all cases).

The optimum area ratio was determined by using the mission/vehicle model developed by Dr. Martin at the University of Alabama and varying the area ratio. The case using 3,000 psi was also examined by LaRC. The results in terms of optimum area ratio were the same. The optimum area ratio was that which resulted in a nozzle exit pressure of 5 psi. (Although the optimum was very flat from 4 to 6 psi.)

The weights were determined using the same procedure as for the bipropellant engine weight validation, i.e., a bottoms up CAD design using the same groundrules, materials, safety factors, limit loads, minimum casting wall thicknesses, etc. Thus, the bipropellant and tripropellant weights are consistent and use the same levels of technology. Figure 77 shows the details for the bipropellant and the tripropellant cases.

The vehicle dry weights were calculated using the LaRC vehicle codes but run by Rocketdyne (after ensuring that Rocketdyne got the same results as LaRC for similar cases). Thus the vehicle results also had consistent assumptions regarding what was or was not included in the engine weights. And these assumptions were consistent with how the engine weights were calculated. The results are shown in Figure 78.

Figures 79 and 80 show the effects of changes to the thermal protection system (TPS) and structure weight factors used in the vehicle weight code. As expected increasing these factors increased the value of bulk density and thus favored the tripropellant concepts.

Summary. Figure 81 summarizes the results of this study. The study has shown that the SSTO application is reasonable based on near to midterm engine technologies and that both bipropellant and tripropellant engines are viable approaches. The chamber pressures needed are ~4,000 psi but the turbine temperatures can be much lower than the current SSME experience. Consequently, it should be possible to achieve the needed chamber pressures while maintaining durability.

Certain technologies were identified as important to achieving the results of this study. Figure 82 shows the implications of the various technologies used on the results of the study. Interestingly, all the technologies identified apply to both bipropellant and tripropellant engines.

Documentation. Results of the Advanced Low-Cost Engine task were included in the final program review, Option 1, briefing book (3 May 1994). A detailed Task Final Report has also been submitted as a separate document. All of these are available from MSFC/PD. Also two AIAA papers, 94-2950 and 94-3317, have been presented showing the results of this task.

Tripropellant Comparison Study

Recent evaluations of main engine options for the SSTO/RLV mission have considered the use of a tripropellant fueled engine to take advantage of its improved fuel density. A wide range of options have been addressed including configuration, technology level, and design practices. A study was conducted to evaluate engine configurations on a consistent basis of technology level, design practice, and design groundrules. Engine weight and performance of several cycles and cycle variants were determined and then compared on a vehicle dry weight basis to determine the merit of each. A series of bipropellant configurations were also studied under identical groundrules to evaluate the inherent differences due to propellant selection only. The results showed that the three major variants considered, a single-chamber tripropellant, an annular bell tripropellant and a bipropellant, were nearly identical in terms of overall vehicle performance.

Background. Two recent thrusts have rekindled the community's interest in the prospects for an efficient single-stage-to-orbit (SSTO) transportation system: the NASA Access to Space Study and the SSTO demonstration program leading up to the DC-X flight program. These efforts and their successors in the reusable launch vehicle (RLV) and X-33 have examined a wide spectrum of liquid rocket engine designs, candidates and technologies which could meet the challenging performance and economic requirements for enhanced access to space. The space transportation community has focused on two major options - an oxygen/hydrogen bipropellant engine and an oxygen/hydrogen/kerosene tripropellant engine. Both options use elements of U.S. engines like the SSME, of foreign engines like the RD-701, and of maturing technologies yet to be flight-proven.

A study was conducted, under Marshall Space Flight Center contract NAS8-39210, to evaluate future engine concepts using tripropellants for the SSTO mission. The study objective was to provide an unambiguous comparison among various tripropellant implementation approaches and cycle choices, and then to compare them to similarly designed bipropellant engines in the SSTO mission. Consequently, the study was based on a "clean sheet" engine design approach. Each concept was optimized as a new engine for the SSTO mission and used consistent design groundrules including design practices and technologies. The study did not try to accommodate specific existing engines or components although they may have utility for reasons of cost or schedule.

The major objective was performance focused and was selected to uncover the inherent advantages of various options. An additional objective was to produce engine concepts which, besides reducing vehicle life cycle costs through decreasing vehicle empty weight, would also lower engine costs through component and operating parameter choices, the inclusion of specific technologies, and improved engine operability.

Study Groundrules. The baseline mission examined for the evaluation of main engine candidates was the reference used in the Option 3 of the Access to Space Study, i.e., an International Space Station (ISSA) resupply to 220 nmi at 51.6 degrees from the KSC launch site. Payload was set at 25,000 pounds. A vertical takeoff, horizontal landing winged body configuration was used. The vehicle technologies employed were consistent with those used in the Access Study (e.g., Li-Al tanks).

To develop the overall performance figure of merit, vehicle dry weight, the methodology used in the Access Study was also employed. Trajectories were flown using the industry-standard POST code using aerodynamic coefficients supplied by NASA-LaRC. The vehicle model employed was the NASA-LaRC CONSIZ code. The 5/94 version of CONSIZ for the winged body vehicle was used since it provided the most consistent comparison between bipropellant and tripropellant vehicle designs. A correction to CONSIZ was also made to account for the differing amounts of helium seal purges required by the engine turbomachinery and drive design options. Engine thrust was determined by setting the vehicle thrust-to-weight at liftoff to 1.2, consistent with the Access Study.

Each engine option considered was optimized to provide lowest vehicle dry weight. Among the factors included were tripropellant engine thrust split between Mode 1 (all three propellants being utilized) and Mode 2 (oxygen/hydrogen only), the amount of mode 1 hydrogen, mixture ratio (MR), nozzle exit pressure, and turbomachinery/drive configuration. The bipropellant was also optimized for MR, nozzle expansion, and configuration.

Weights for each option were generated by using detailed CAD engine design layouts at selected points and scaling from those points. The buildup of a complete engine option weight statement was based on results from the CAD 3-D evaluation, on results from SSME and kerosene engine auxiliary components, and from Space Transportation Main Engine (STME) design details. The resulting weights include all component categories and elements contained in the SSME weight statement.

A key groundrule for this evaluation was consistency across the spectrum of engine designs. This was applied in the area of design groundrules and practices in areas such as safety factors, material selection, and fabrication processes. Equally important was the application of equivalent technology level across the designs - Technology Readiness Level (TRL) 6 by 2002. These constraints permit an "apples-to-apples" contrast of options on their inherent performance merit rather than by accident of their organizational origin and design year.

Engine Options. Engine options were defined in terms of cycle and particular component configuration or arrangement within the cycle. Two bracketing methods of tripropellant implementation were examined: a single main combustion chamber (MCC) wherein all three propellants were burned in Mode 1 with the kerosene shut down for Mode 2; and an annular arrangement where the oxygen and hydrogen were burned in an inner chamber and the oxygen and kerosene burned in a ring of outer chambers attached downstream of the inner chamber throat. Figure 83 depicts the two configurations. The bipropellant options used a conventional single MCC configuration. All options used a fixed-position bell nozzle.

The study defined five basic engine cycles for inclusion in the tripropellant evaluation. These included a full flow staged combustion cycle (FFSCC) (specifically a mixed preburner configuration), a fuel-rich staged combustion cycle (FRSCC), an oxidizer-rich staged combustion cycle (ORSCC), a hybrid staged combustion cycle (SCC), an expander cycle, and a gas generator cycle. In addition, various arrangements of turbopumps and preburners were studied to evaluate engine weight, performance, and operating parameters. These are illustrated in Figure 84.

A multitude of possible tripropellant configurations were examined and led to the selection of leading candidates for the rest of the study. These are summarized in Figure 85 for a nominal chamber pressure of 4,000 psia and 421,000 pounds sea-level thrust. The configuration selection was based primarily on engine weight and turbine operating temperatures.

Similarly, the study considered the comparative merits of bipropellant oxygen/hydrogen engines. The previous task, Advanced Low-Cost Engines, showed that the FFSCC, the fuel-rich SCC, and the hybrid cycles were robust, high performance options for the SSTO application. The previous work was updated to be rigorously consistent with the tripropellant concepts in terms of design and technology content. Their characteristics are included in Figure 85.

Concept Performance. The engine overall performance design factors were optimized for minimal vehicle dry weight over the pressure range of interest for the SSTO mission, 2,000 to 4,500 psia. Figure 86 summarizes the baseline selection. The transition point in the trajectory from Mode 1 to Mode 2 was also optimized for each design point.

The exit pressure optimization was conducted at 4,000 psia chamber pressure and used at other chamber pressures to determine nozzle expansion ratios. The previous task, Advanced Low-Cost Engines, showed that the optimal nozzle exit pressure is flat over the pressure range for the SSTO mission. Figure 87 depicts the nozzle exit pressure trends for the three basic configurations - bipropellant, single chamber tripropellant, and annular bell tripropellant. Engine cycle configuration details were not a significant factor in the trends.

The nozzle exit pressure evaluation is based on the trade of increasing mission specific impulse (Isp) against increasing engine weight as pressure drops (or, as nozzle area ratio increases). The FFSCC is shown but evaluations for the other cycles considered showed no significant differences. An optimum exit pressure always exists since as the exit pressure is increased (area ratio is decreased) less and less nozzle weight is saved because the amount of surface area of nozzle per increment of area ratio decreases but the amount of specific impulse lost increases per increment of area ratio. Consequently, the weight decrease gets small while the specific impulse loss gets large. On the other hand, at high area ratios the change in nozzle weight per increment of area ratio gets very large while the gain in specific impulse gets very small.

All the cases shown in Figure 87 have the same characteristics: a sharp improvement as exit pressure decreases up to a point, but then a flatness around the optimum. Two cases are shown for the bipropellant at two significantly different mixture ratios (6.0 and 6.9) to show that the exit pressure optimum is not sensitive to mixture ratio.

The mixture ratio evaluation trade for the oxygen/hydrogen propellant operation for the three configurations is illustrated in Figure 88 for the FFSCC. The driving trade is, of course, lower tank weight as propellant density increases versus the increase in tank weight due to higher propellant requirements at lower performance levels. A secondary factor is a slight decrease in engine weight as MR is increased. The bipropellant optimizes at near 6.9. The annular bell tripropellant optimizes near the same MR because of the largely independent operation of the oxygen/hydrogen and oxygen/kerosene sections of the engine. One non-independent feature of its operation is the use of hydrogen to cool both engine sections. The engine optimizes at a thrust split between the oxygen/kerosene and oxygen/hydrogen sections set by the hydrogen cooling limit (1000°R bulk temperature for this study). This cooling limit occurs at different thrust splits versus MR. Designing along the cooling limit line slightly lowers the optimum MR and decreases the sensitivity to MR versus designing along a constant thrust split line. The net effect is to slightly lower the optimum MR to about 6.8.

The single chamber tripropellant is much more integrated in its overall operation and tends to optimize near a MR of 6. The engine weight of the single chamber configuration is set by Mode 1 operation. The mixture ratio in Mode 2 only affects the off-design operating point. Consequently, the effect of Mode 2 MR is confined to bulk density versus specific impulse effects, and that only in the upper part of the trajectory. The effects are to favor specific impulse more strongly than the engines which utilize oxygen/hydrogen operation throughout the entire trajectory, i.e., the optimum MR is lower (6.2), and the sensitivity to MR is much less.

Similar analyses were conducted for the oxygen/kerosene Mode 1 operation MR. Vehicle weight was much less sensitive to MR excursions because the bulk density of oxygen versus varying amounts of kerosene with a fixed percentage of hydrogen varies much more gently than oxygen versus hydrogen alone. Also the engine weight increases slightly as this MR is increased (versus an engine weight decrease as the oxygen/hydrogen MR is increased). The net effect is that vehicle dry weight is flat across an oxygen to kerosene plus hydrogen mixture ratio of 4.0 to 4.6 (less than 0.2% difference).

An evaluation of the effect of engine cycle type on vehicle dry weight as a function of chamber pressure was conducted. In all instances, optimum design and flight parameters were used at each design point in order to derive meaningful comparisons between options.

Figure 89 depicts the dry vehicle weight performance of the single chamber tripropellant design over the 2,000 to 5,000 psia range. Because all the cycles illustrated are closed cycles, the performance levels are equal for each and the small differences shown reflect only the engine weight trends. The data shows little variation among the cycles, on the order of 3%, across the range of pressures. Recall that these pressure levels are achieved only during the Mode 1 operation. During Mode 2 the pressure drops, for example, from 4000 psia to 1960 psia as the kerosene and part of the oxygen flows are shut down.

The annular bell tripropellant cycle comparison is shown in Figure 90. It is assumed that the hydrogen and the kerosene sections of the engine operate at the same chamber pressure and use the same cycle. Again the data shows small weight differences (<5% at low pressures, declining to 1% as pressure increases) among the cycles.

The bipropellant engine cycle comparison is shown in Figure 91. This engine operates at a constant condition throughout the entire flight profile. The analysis results show vehicle weight differences less than 2% among the closed cycle options shown. A gas generator cycle was also evaluated but is shown in Figure 91 to yield 8% higher vehicle weights due to the low performance of the open cycle engine despite the fact that the engine was about 11% lighter than the closed cycle options.

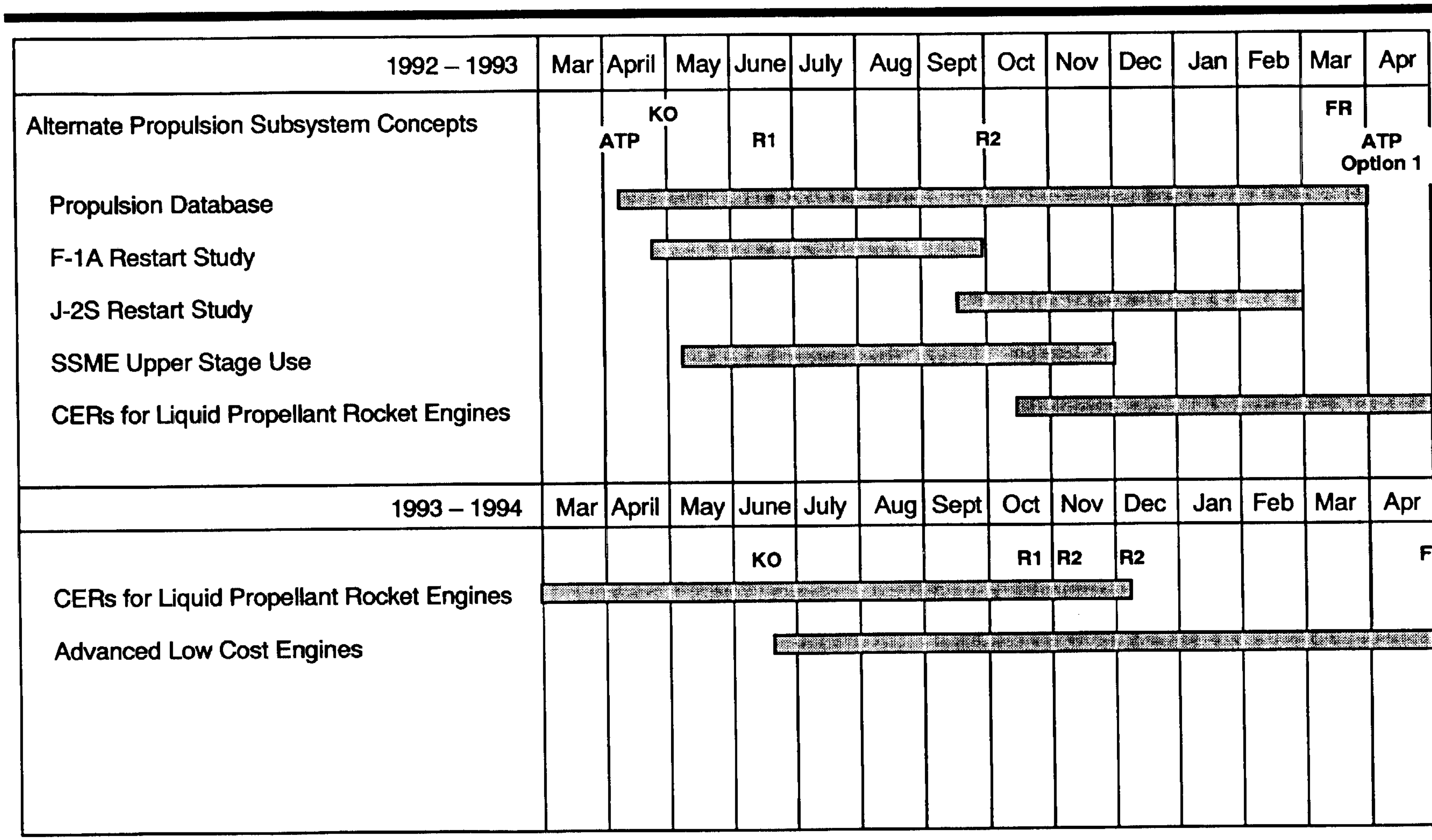
For purposes of comparison, the three baseline configurations are shown in Figure 92 for the FFSCC. The comparison shows that the weight difference at the vehicle level is about 2% at 4000 psia and within 3% throughout the chamber pressure range.

Summary. The results of this evaluation have shown that, when compared on a consistent basis, performance is not an inherent discriminator in the selection among bipropellant and tripropellant options for the SSTO mission. Previous studies have focused on available engines, non-optimized engines, or engines designed to different groundrules in evaluating the bipropellant vs. tripropellant issue and have not uncovered the performance difference inherent in the propellant selection alone.

The results of the study have also shown that the vehicle weight performance differences driven by engine parameter selections such as MR, chamber pressure, and so on are much more important than the bipropellant/tripropellant selection. Other factors not addressed here, such as the effect of margins and the use of uncoated materials, also may be a much larger factor in the trades of cycle and propellant selection.

Documentation. Results of the Tripropellant Comparison Study task were included in a detailed Task Final Report has also been submitted as a separate document. All of these are available from MSFC/PD. Also two AIAA papers, 94-4676 and 95-3609, have been presented showing the results of this task.

Advanced Transportation System Studies
Technical Area 3
Alternate Propulsion Subsystem Concepts



ATP – Authority to Proceed
KO – Kick-Off Meeting
R# – Interim Review #
FR – Final Review

TA3-0001m2

Figure 1. Contract Schedule

Meeting	Date	Location	Topics	Documentation
Orientation Meeting	29 April 1992	MSFC	Study Plan	Briefing Book (DR-2) Study Plan (DR-1)
Formal Study Review	17 June 1992	MSFC	F-1A Restart Study SSME Upper Stage Use Database Development	Briefing Book (DR-2)
Informal Study Review	20 July 1992	Rocketdyne	F-1A Restart Study SSME Upper Stage Use	–
Informal Study Review	1 September 1992	Rocketdyne	F-1A Restart Study Planning for Future Study Tasks	–
Formal Study Review	1-2 October 1992	MSFC	F-1A Restart Study (Task Final Review) SSME Upper Stage Use Database Development	Briefing Book (DR-2) F-1A Restart Study Task Final Report (Part of DR-4)
Informal Study Review	21 October 1992	Rocketdyne	F-1A Restart Study SSME Upper Stage Use	–
Formal Study Review	17 March 1993	MSFC	J-2S Restart Study (Task Final Review) SSME Upper Stage Use (Task Final Review) Database Development (Review of First Year Work) CERs for Liquid Engines	Briefing Book (DR-2) J-2S Restart Study Task Final Report (Part of DR-4) SSME Upper Stage Use Task Final Report (Part of DR-4) Database Development Task Final Report (Part of DR-4) Final Report Volumes I, II, and III (DRs 4, 5, 6, 8, 9)

TA3–0409aFig

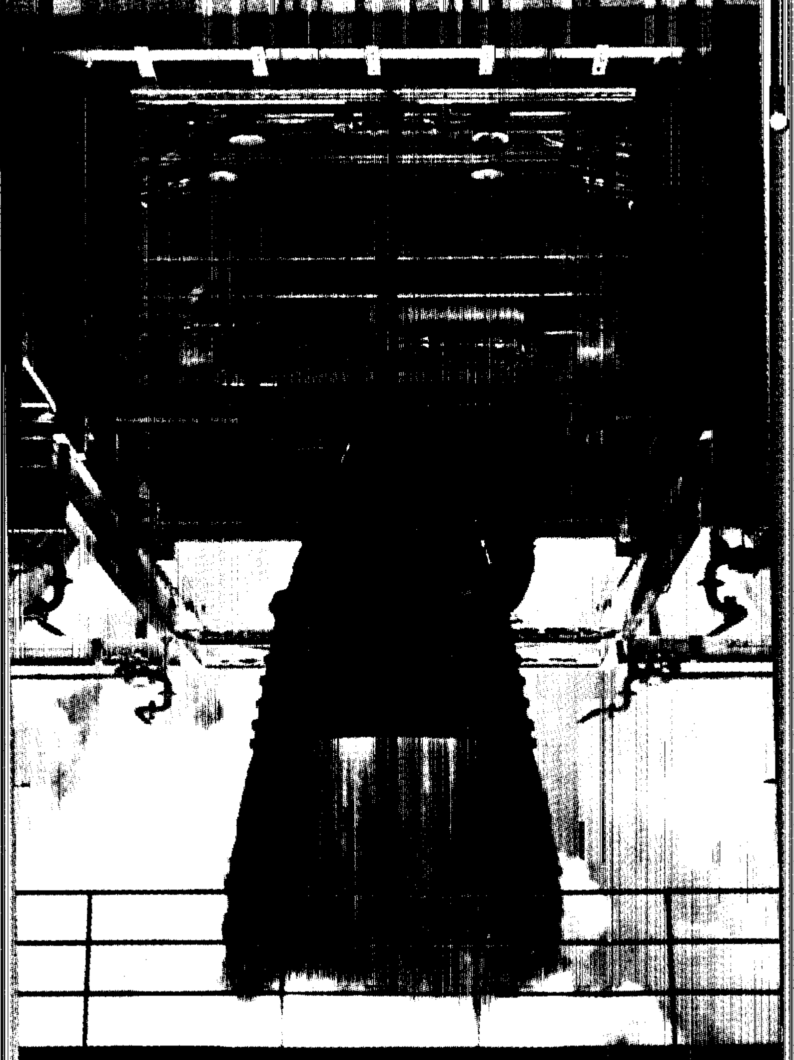
Figure 2. Reviews and Documentation

Meeting	Date	Location	Topics	Documentation
Orientation Meeting	22 June 1993	MSFC	Study Plan	Briefing Book (DR-2)
Formal Study Review	12 October 1993	MSFC	Advanced Low Cost Engines	Briefing Book (DR-2)
Informal Study Review	21 October 1993	LaRC	Advanced Low Cost Engines	–
Informal Study Review	22 October 1993	NASA-HQ	Advanced Low Cost Engines	–
Formal Study Review	9 November 1993	MSFC	Advanced Low Cost Engines	Briefing Book (DR-2)
Formal Study Review	9 December 1993	MSFC	CERs for Liquid Engines (Final Task Review)	CERs for Liquid Engines Task Final Report (Part of DR-4)
Informal Study Review	17 March 1994	LeRC	Entire Contract	–
Formal Study Review	3 May 1994	MSFC	Advanced Low Cost Engines (Task Final Review)	Briefing Book (DR-2) Advanced Low Cost Engines Task Final Report (Part of DR-4) Final Report Volumes I, II, and III (DRs 4, 5, 6, 8, 9)

TA3–0409bFig

Figure 2. Reviews and Documentation (Cont'd)

F-1/F-1A Engine Performance



• Performance & weight	F-1	F-1A
• Nominal thrust (lb/f)		
• Sea level	1,522,000	1,800,000
• Vacuum	1,748,200	2,020,700
• Specific impulse (sec)		
• Sea level	265.4	269.7
• Vacuum	304.1	303.1
• Chamber pressure (psia) (nozzle stagnation)	982	1,161
• Engine mixture ratio	2.27	2.27
• Expansion ratio	16:1	16:1
• Dimensions		
• Length (in.)	220.4	220.4
• Diameter (in.)		
• Powerhead	104.75	104.75
• Exit	143.5	143.5
• Weight (lb)	18,616 **	19,000 (est*)

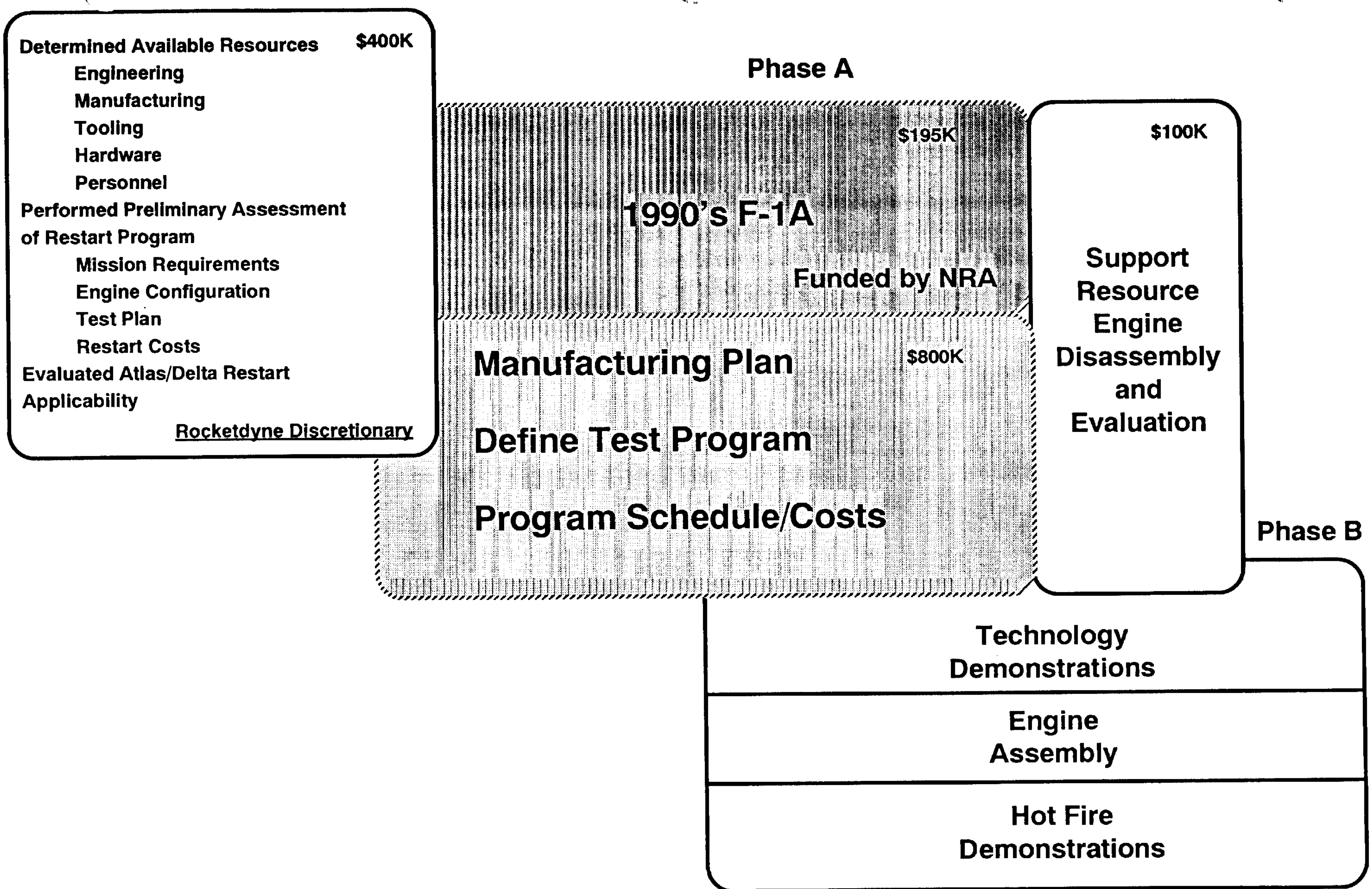
* Includes accessories – interface panel, heat exchanger & fluid, instrumentation, insulation, pressurization ducts, & throttling system

** Engine F-2090 & subs

Rockwell International
Rocketdyne Division

SC91c-12-1054B
(1DB24-2/16/85-E1*) M469

Figure 3.



TA3–0228Fig

Figure 4. F-1A Restart Program Overview

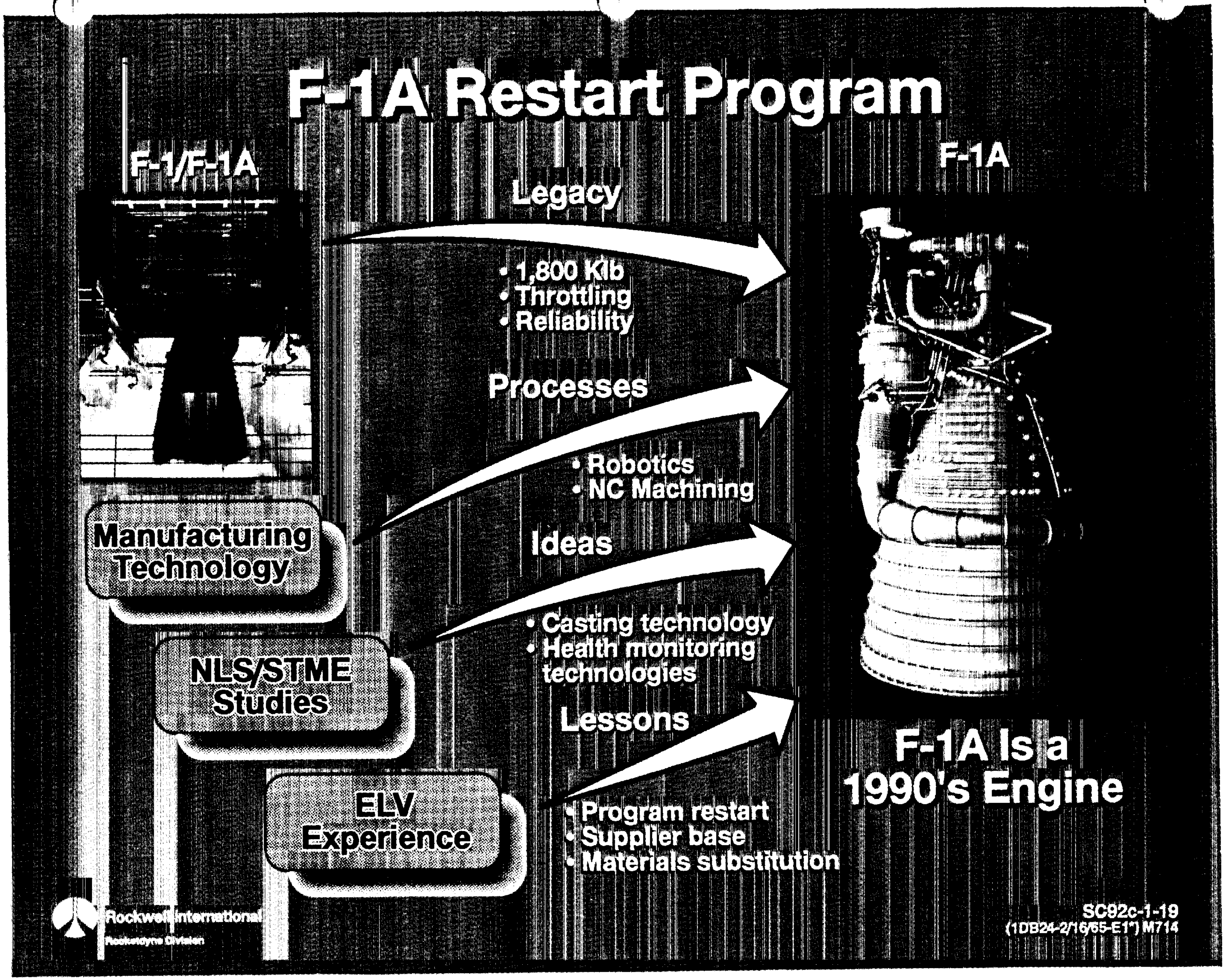
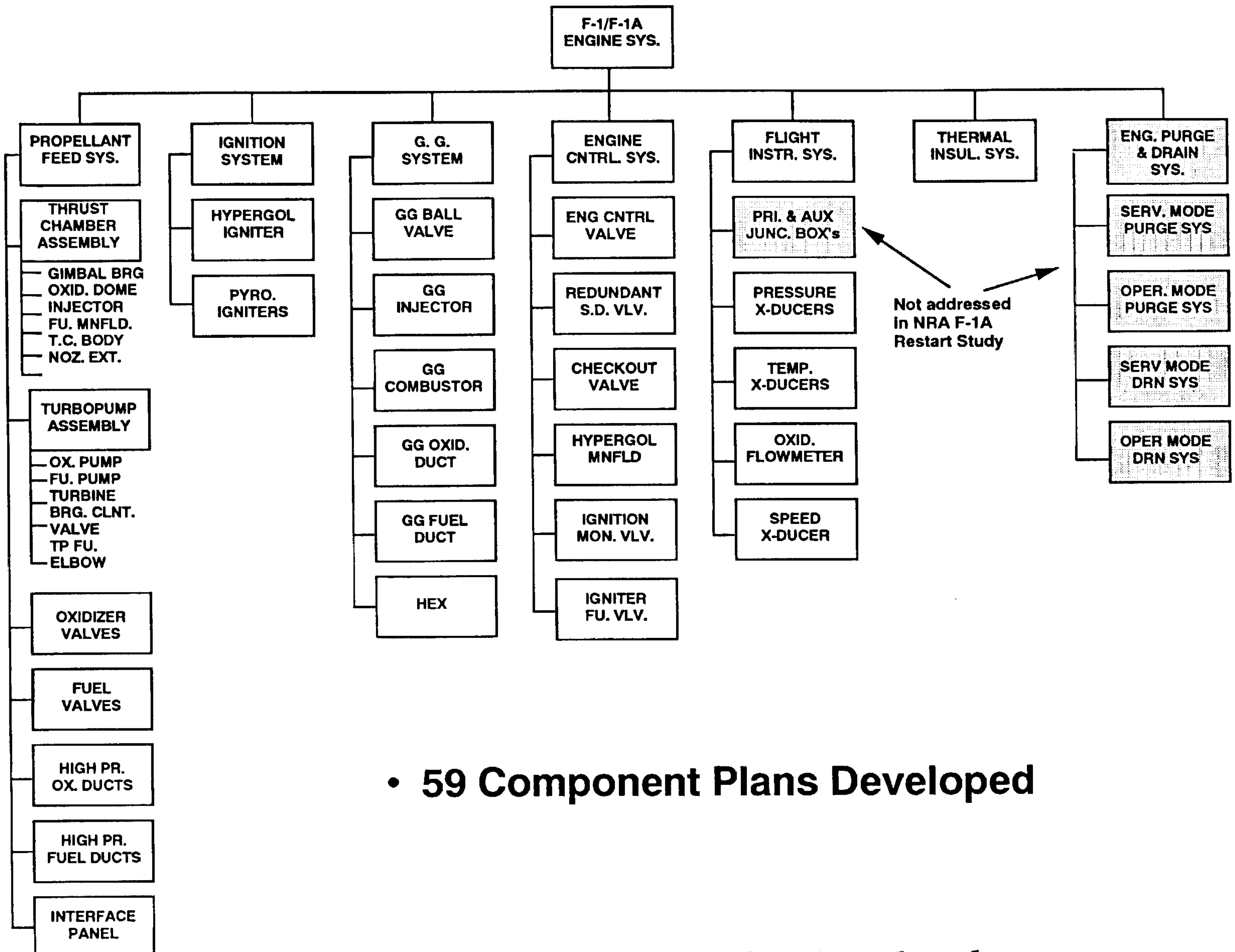

F-1A Restart Program
F-1/F-1A
F-1A
Legacy
• 1,800 Klb
• Throttling
• Reliability
Processes
• Robotics
• NC Machining
Ideas
• Casting technology
• Health monitoring technologies
Lessons
• Program restart
• Supplier base
• Materials substitution
Manufacturing Technology
NLS/STME Studies
ELV Experience
F-1A Is a 1990's Engine
Rockwell International
Rocketdyne Division
SC92c-1-19
(1DB24-2/16/65-E1") M714

Figure 5.

F-1A Restart Study

F-1A Engine Components Evaluation



- **59 Component Plans Developed**

Figure 6. Component Plans Developed

TA3-0191Fig

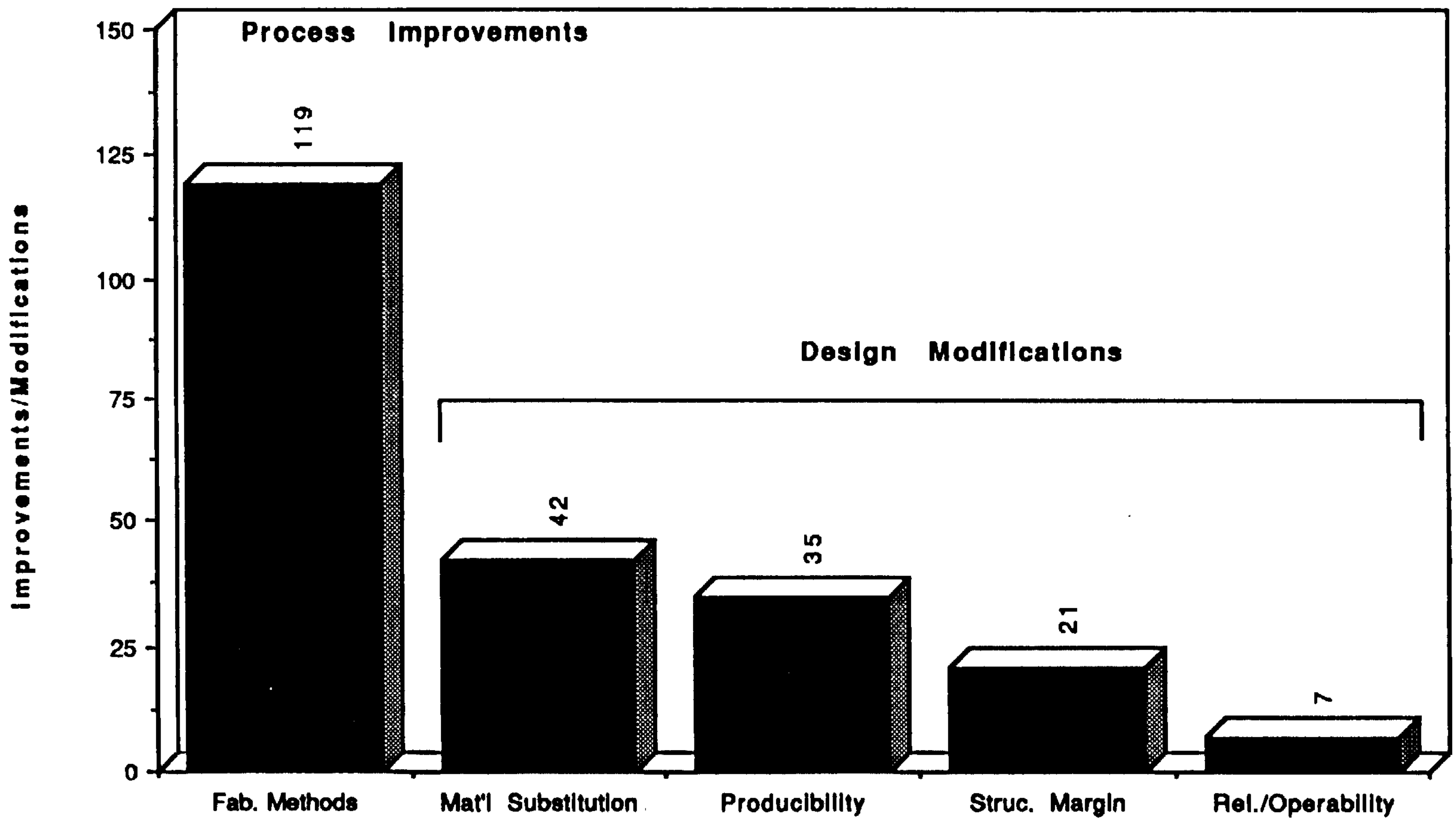


Figure 7. F-1A Component Plan Summary

TA3–0219Fig

F-1A RESTART STUDY
STATE-OF-THE-PRACTICE FABRICATION METHODS

Component \ Fabrication Method	ROBOTICS	NUMERICAL CONTROL MACHINING	LASER OPERATIONS	CASTING IMPROVEMENTS	FORGING IMPROVEMENTS	MACHINE CELL	ALTERNATE TOOLING
MAIN INJECTOR		✓	✓			✓	
GIMBAL		✓			✓		
LUGS		✓		✓	✓		
GG DUCTS	✓	✓					
TURB. BLADES		✓		✓			
2ND STG. NOZ.		✓		✓			
FUEL INDUCER		✓		✓			
TP FUEL INLET		✓		✓		✓	✓
IMPELLERS		✓		✓			
TP OX. INLET		✓		✓			✓
DUCTS/ LINES (15)	✓	✓					
MOV's		✓				✓	
MFV's		✓				✓	

119 Improvements Identified in 59 Component Plans

Figure 8. State-of-the-Practice Fabrication Methods

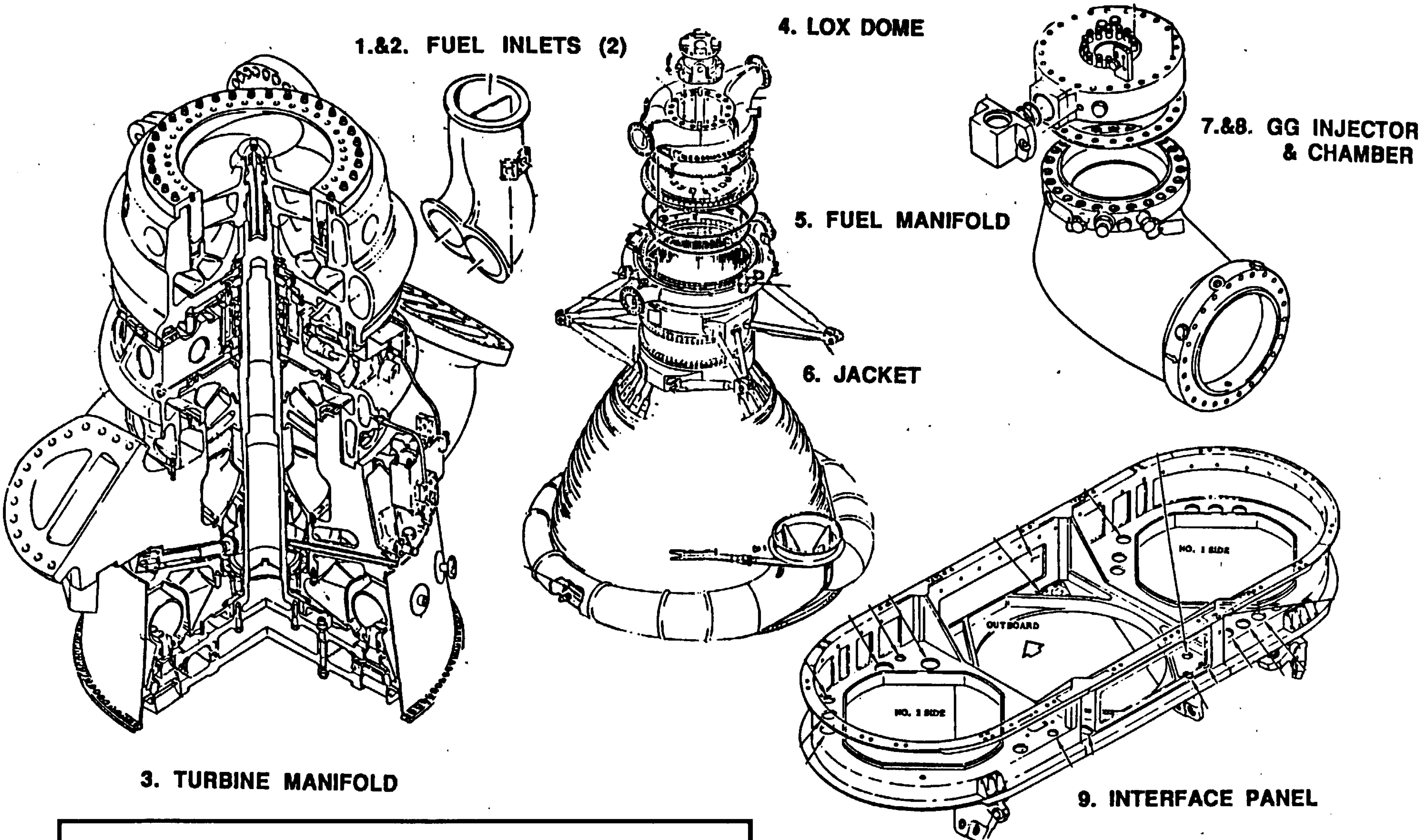
TA3–0193Fig

F-1A RESTART STUDY MATERIAL SUBSTITUTION			
COMPONENT \ SUBSTITUTION DRIVER	ENVIRONMENT	OBSOLESCENCE	STRESS CORROSION SUSCEPTIBILITY
DOME/JACKET/TUBES		Inconel X 750 to Inconel718	
GIMBAL BEARING	Cadmium Plating		
NOZZLE EXTENSION	Asbestos seals	Hastelloy C	
GG CHAMBER		Hastelloy C to Inconel 625	
IMPELLERS	TENS 50 (Berylium) to A356/357		
VOLUTES	TENS 50 (Berylium) to A356/357		
FUEL/LOX INLET ASSY's	TENS 50 (Berylium) to A356/357		
FUEL INLET ELBOWS	TENS 50 (Berylium) to A356/357		
DUCTS & LINES (13)			321 CRES Bellows Inconel 625
PROP. & CONTROL VLV's			17-7PH, 2024 Al, 7075 Al, AM355
GG BALL VALVE	TENS 50 (Berylium) to A356/357		
PYROTECHNIC IGNITERS	Radioactive diode	Radioactive diode	
THERMAL BLANKET	Asbestos		

42 Modifications Identified in 59 Component Plans

TA3-0194Fig

Figure 9. Material Substitution Recommendations



9 Casting Conversions Planned

Figure 10. Casting Conversions

TA3–0215Fig

F-1A Restart Study
F-1A 1800K Engine Structural Margin

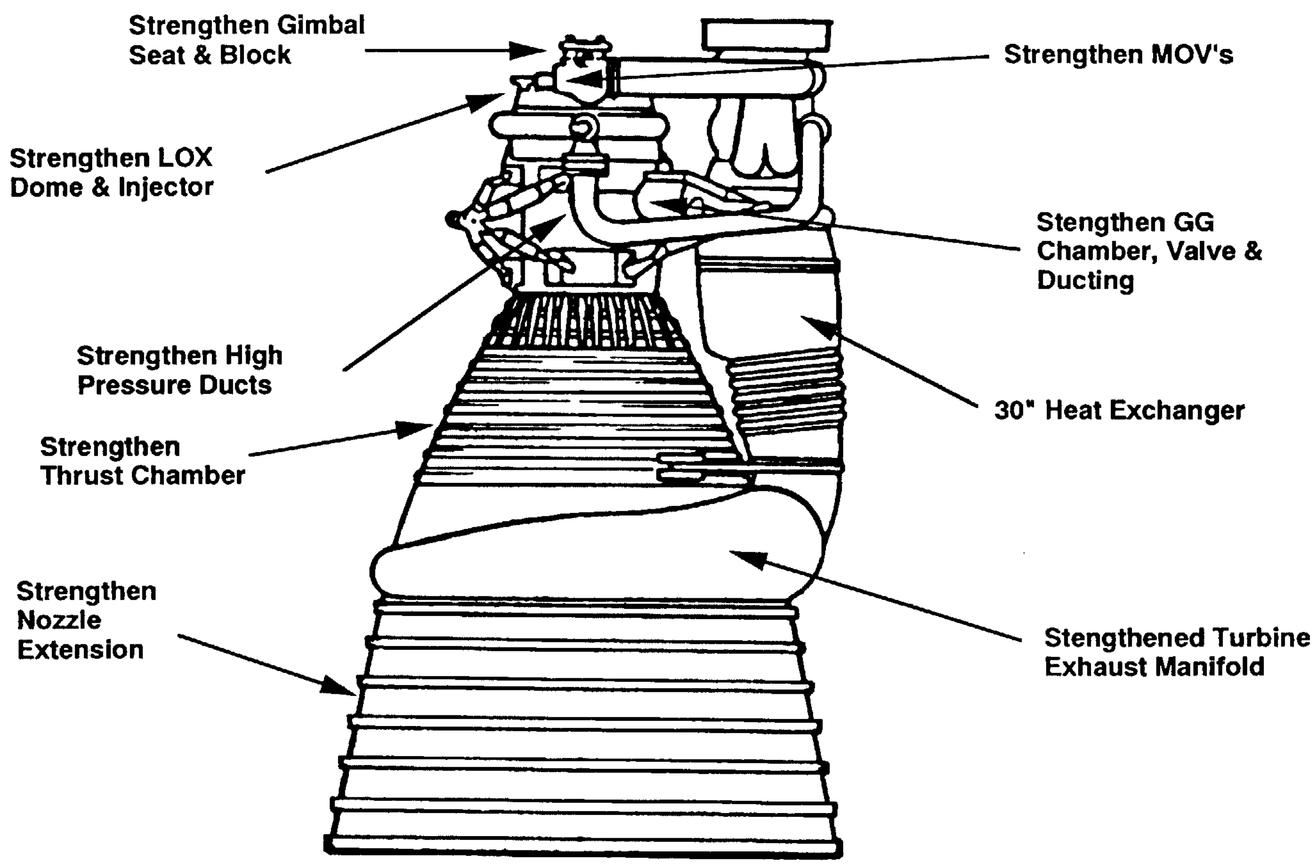


Figure 11. Recommended Structural Margin Modifications

TA3–0217Fig

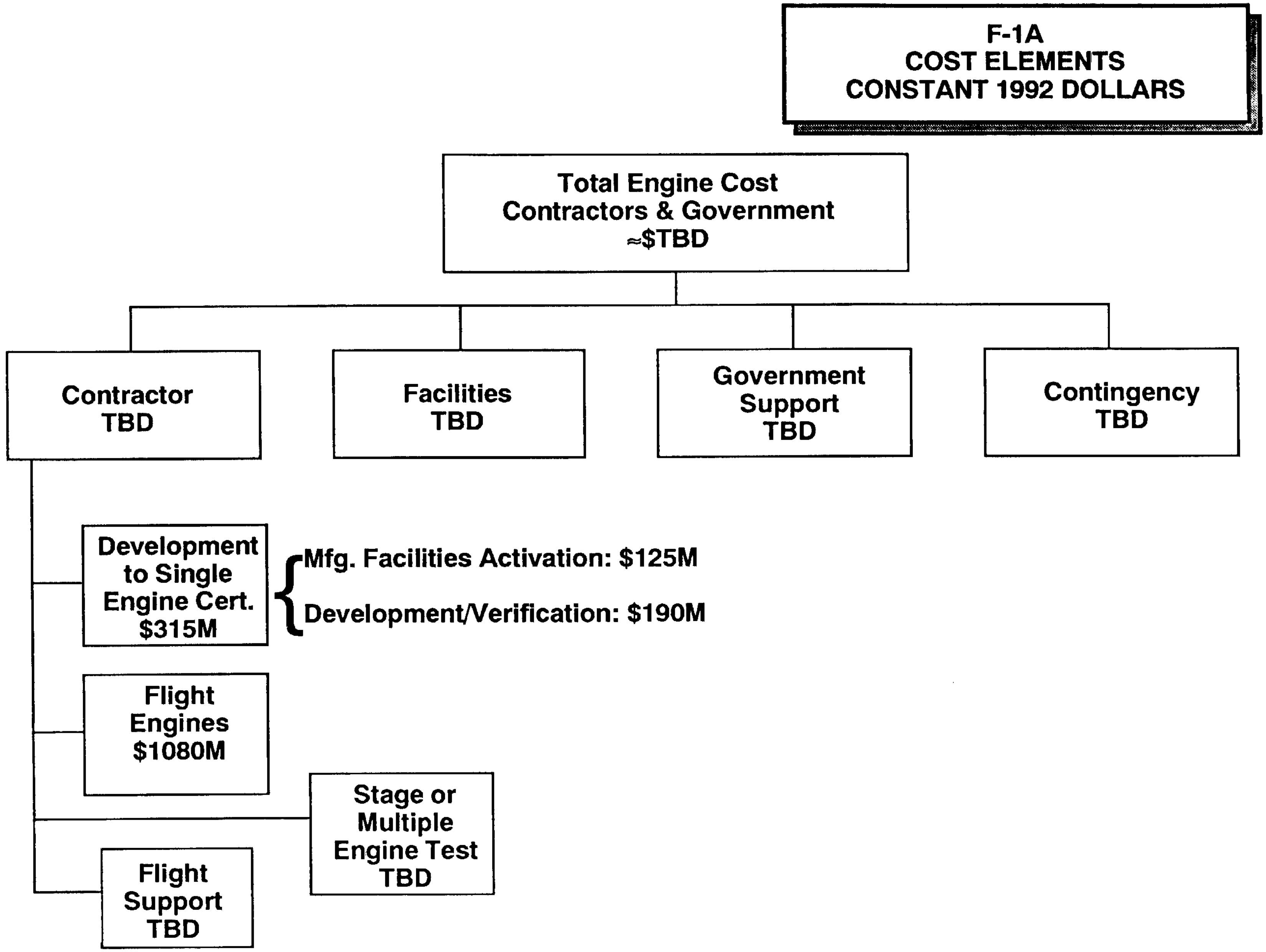


Figure 12. F-1A Restart Program Cost Elements

TA3–0230Fig

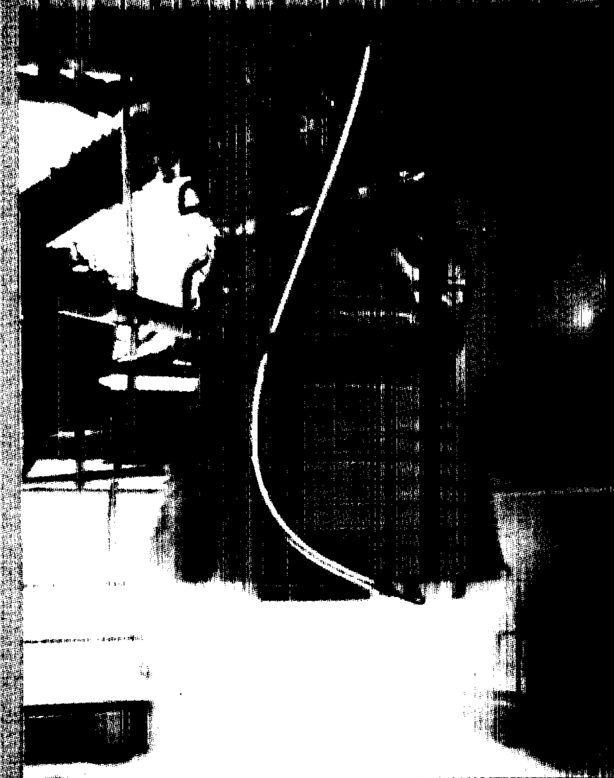
J-2S Basic Engine Features

Performance & Weight

• Nominal vacuum thrust (lb)	265,000
• Nominal vacuum specific impulse (sec)	436
• Chamber pressure (psia) (nozzle stagnation)	1,200
• Engine mixture ratio calibration O/F	5.5:1
• Basic engine dry weight (lb)	3,235
• Engine dry weight (lb) (including accessories)	3,800

Description

- Pump-fed liquid propellant rocket engine
- Propellants – liquid oxygen & liquid hydrogen
- Tap-off turbine drive cycle
- Tubular-wall thrust chamber, regen cooled
- Nozzle area ratio – 40:1
- Separate oxidizer & fuel turbopumps
- Throttling capability
- Low-thrust operating capability



Testing Experience
Production Configuration*

6 Engines, 273 Tests,
30,858 sec

*R&D configuration totalled 10,756 sec

Rockwell International
Rocketdyne Division

SC91c-12-1055
M468

Figure 13.

J-2S Restart Study
Task Flow

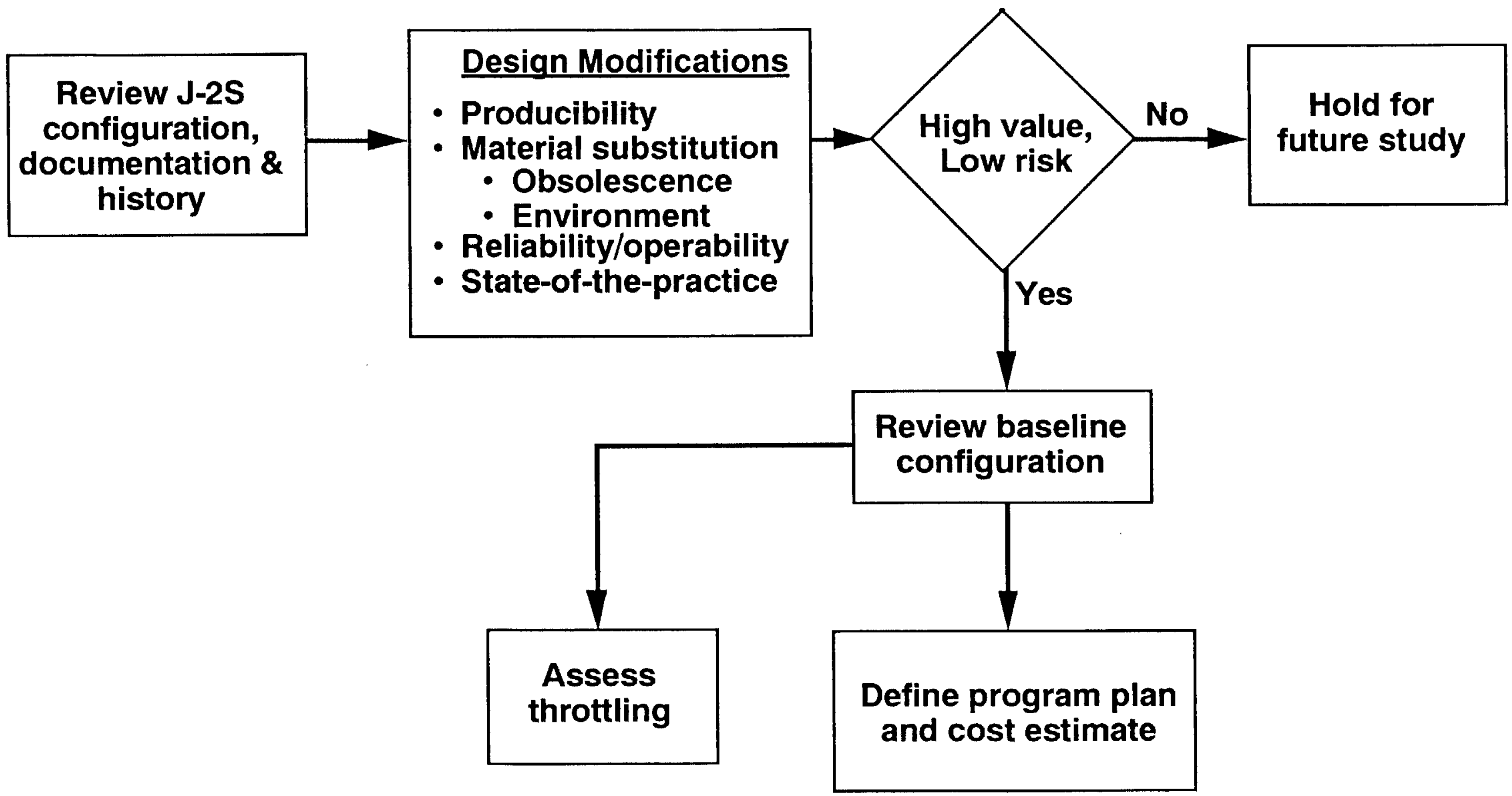


Figure 14. J-2S Restart Study Task Flow

TA3-0385Fig

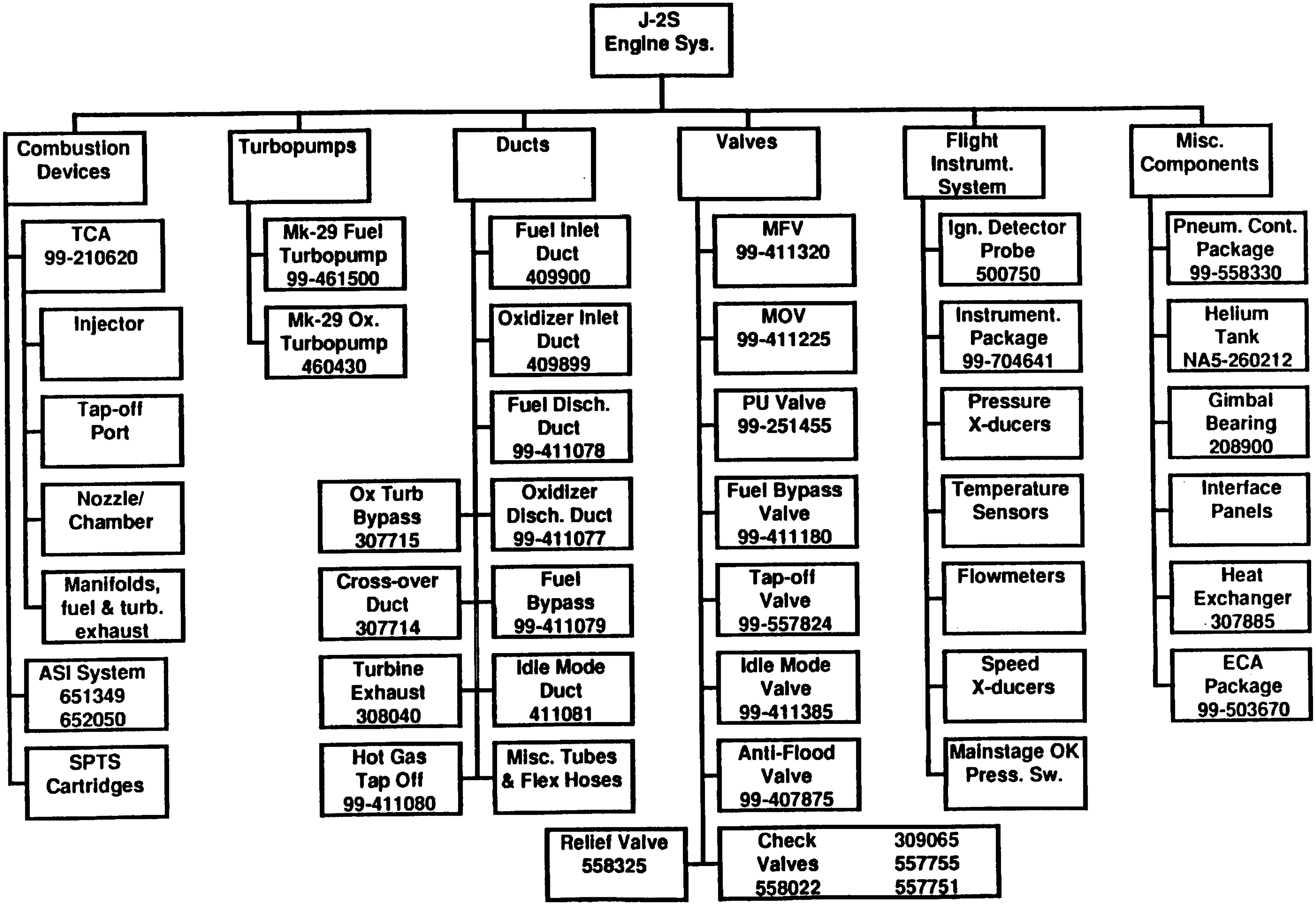


Figure 15. J-2S Engine Components Evaluation

TA3-0389Fig

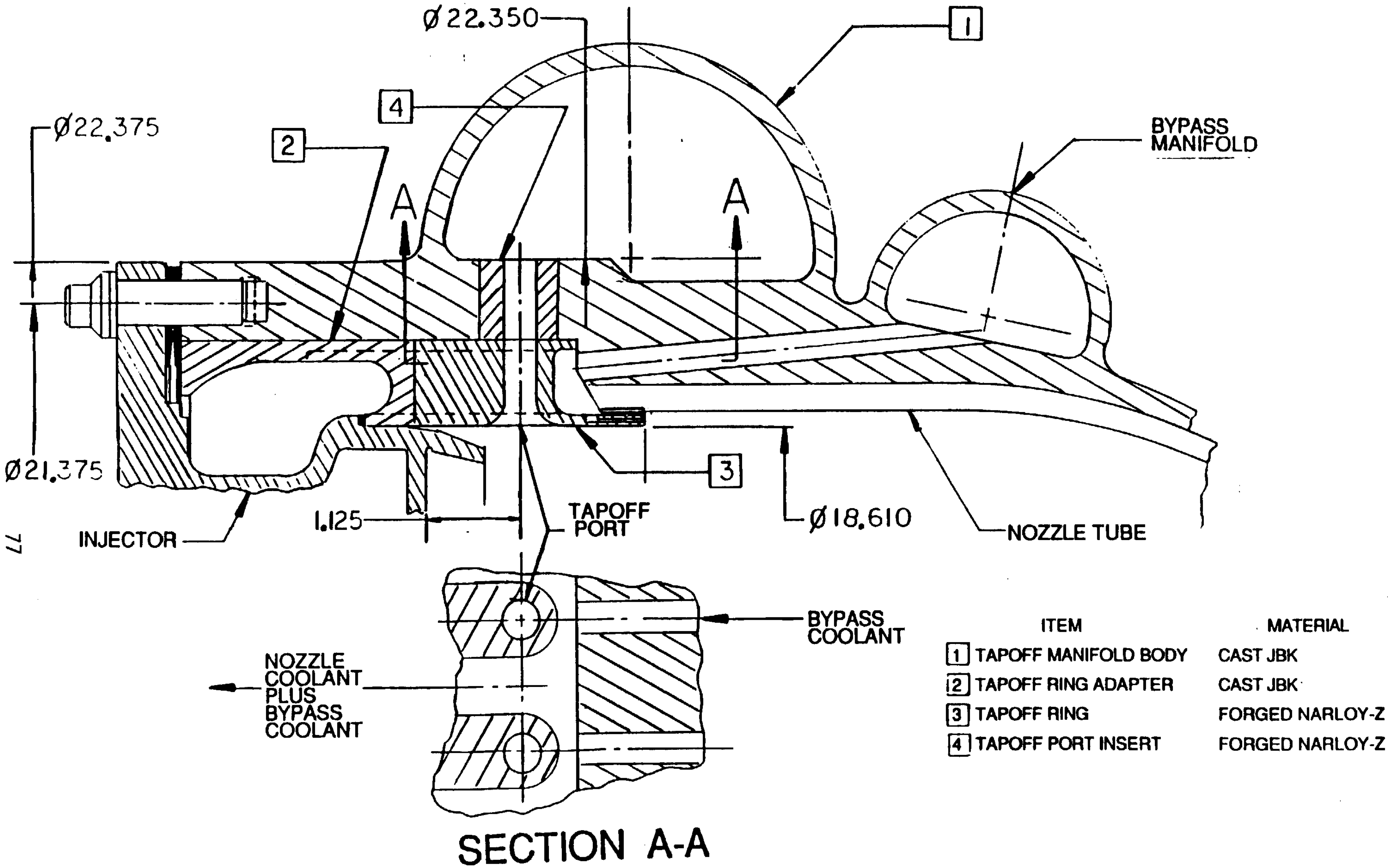


Figure 16. Proposed J-2S Tap-off Manifold Redesign

TA3-0392Fig

Component	Dwg. Number	Changes from basic drawing	Mod Type**
Combustion Devices			
Thrust Cham. Assy	210620		
Injector		Use castings for body to eliminate welds. Braze posts.	1,2
Tap-Off Port		Use casting to replace port, manifolds, & J-Tubes.	2,3
Nozzle/Chamber		Full length nozzle jacket with brazed manifolds.	2
Manifolds		Use castings to replace welded structures.	2
ASI System	651349	Machine and braze assembly.	1
SPTS Cartridges	Spec	Update initiator and propellant	No Change
Turbomachinery			
Mk-29 Fuel T/P	461500	Cast impeller, stiffen & modify brgs., & update dynamic seals	1,2,3,4
Mk-29 Ox T/P	460430	Update dynamic seals	1,4
Ducts			
Fuel In Duct	409900	Redesign completely	*
Ox In Duct	409899	Redesign completely	*
Fuel Disch. Duct	411078	Use precision end point tooling	1
Ox Disch. Duct	411077	Use precision end point tooling	1
Fuel Bypass Duct	411079	Use precision end point tooling	1
Idle Mode Duct	411081	Use precision end point tooling	1
Ox Turb B/P Duct	307715	Use precision end point tooling	1
Cross-over Duct	307714	Use precision end point tooling	1
Turb Exh. Duct	308040	Use precision end point tooling	1
H/G Tap-off Duct	411080	Use precision end point tooling	1
Misc Tubes & Hoses	Various	Reassess routing for specific application	No Change

* Redesign instead.
** 1 – Producibility; 2 – Fabrication Technique; 3 – Material Substitution; 4 – Reliability/Operability Enhancement.
No Change – No recommended change.

Figure 17. Recommended Modifications

Component	Dwg. Number	Changes from basic drawing	
Valves			
Main Fuel Valve	411320	Cast housing, die forged butterfly of 431 CRES	2,3,4
Main Ox. Valve	411225	Cast housing, mylar shaft seals, die forged butterfly of 431 CRES	2,3,4
Prop Util. Valve	251455	Update motor design	4
Fuel Bypass Valve	411180	Cast housing and caps, die forged butterfly of 431 CRES	2,3
Tap-off Valve	557824	Elminate butterfly seal, use die forged parts	2
Idle Mode Valve	411385	Cast housing, mylar shaft seals, cast ball, revised seat	1,2,4
Anti-Flood Valve	407875	Cast housing, redesigned poppet	2
Relief Valve	558325	Die forged parts	2
Check Valve	558022	Weld structure, teflon poppet sleeve	1,4
Check Valve	309065	Redesign	1
Check Valve	557755	Weld structure, teflon poppet sleeve, eliminate filter	1,4
Check Valve	557751	Weld structure, teflon poppet sleeve, eliminate filter	1,4
light Instrumentatio			
Ign. Det. Probe	500750	Eliminated	Eliminated
Instr. Package	704641	Redesign	*
Pressure Sensors	Various	Redesign	*
Temp. Sensors	Various	Redesign	*
Flow Sensors	Various	Redesign	*
Speed Sensors	Various	Redesign	*
Valve Pos x-ducers	Various	Redesign	*
M/S OK Press Switch	NA5-27453	Redesign	*

* Redesign instead.
** 1 – Producibility; 2 – Fab Technique; 3 – Material Substitution; 4 – Reliability/Operability Enhancement.
No Change – No recommended change.

Figure 17. Recommended Modifications (Cont'd)

Component	Dwg. Number	Changes from basic drawing	
llaneous Component			
Pneum Cont. Pack.	558330	Replace He tk vent w/ larger valve, add filter, cast housings	2,4
Helium Tank	NA5-260212	Replace with in production unit	1
Gimbal Bearing	208900	Eliminate alignment mechanism	4
Interface Panels	Various	Redesign for specific application	No Change
Heat Exchanger	307885	No changes	No Change
ECA Package	503670	Redesign with modern electronics	*

* Redesign instead.
** 1 – Producibility; 2 – Fab Technique; 3 – Material Substitution; 4 – Reliability/Operability Enhancement.
No Change – No recommended change.

Figure 17. Recommended Modifications (Cont'd)

- Feasibility of J-2S production restart demonstrated
  - No technical issues found
  - Could be produced to existing drawings
    - Only engine electronics absolutely require replacement
    - No material changes required
    - All processes are still possible

- Significant cost reduction potential identified
  - 24 Changes for producibility (no change in form/fit/function)
  - 20 Changes in fabrication technique (i.e., castings, die forging, etc.)

- 23 Other desirable changes identified
  - 12 Material substitutions
  - 11 Reliability/operability enhancements

- All recommended changes are prudent risks
  - Mitigated by ELV experience
  - Justified by benefits
  - Verification test impact within scope of existing plan

J-2S restart production feasibility demonstrated

Figure 18. J-2S Restart Study Conclusions

Rockwell International
Rocketdyne Division

TA3-0401bFig

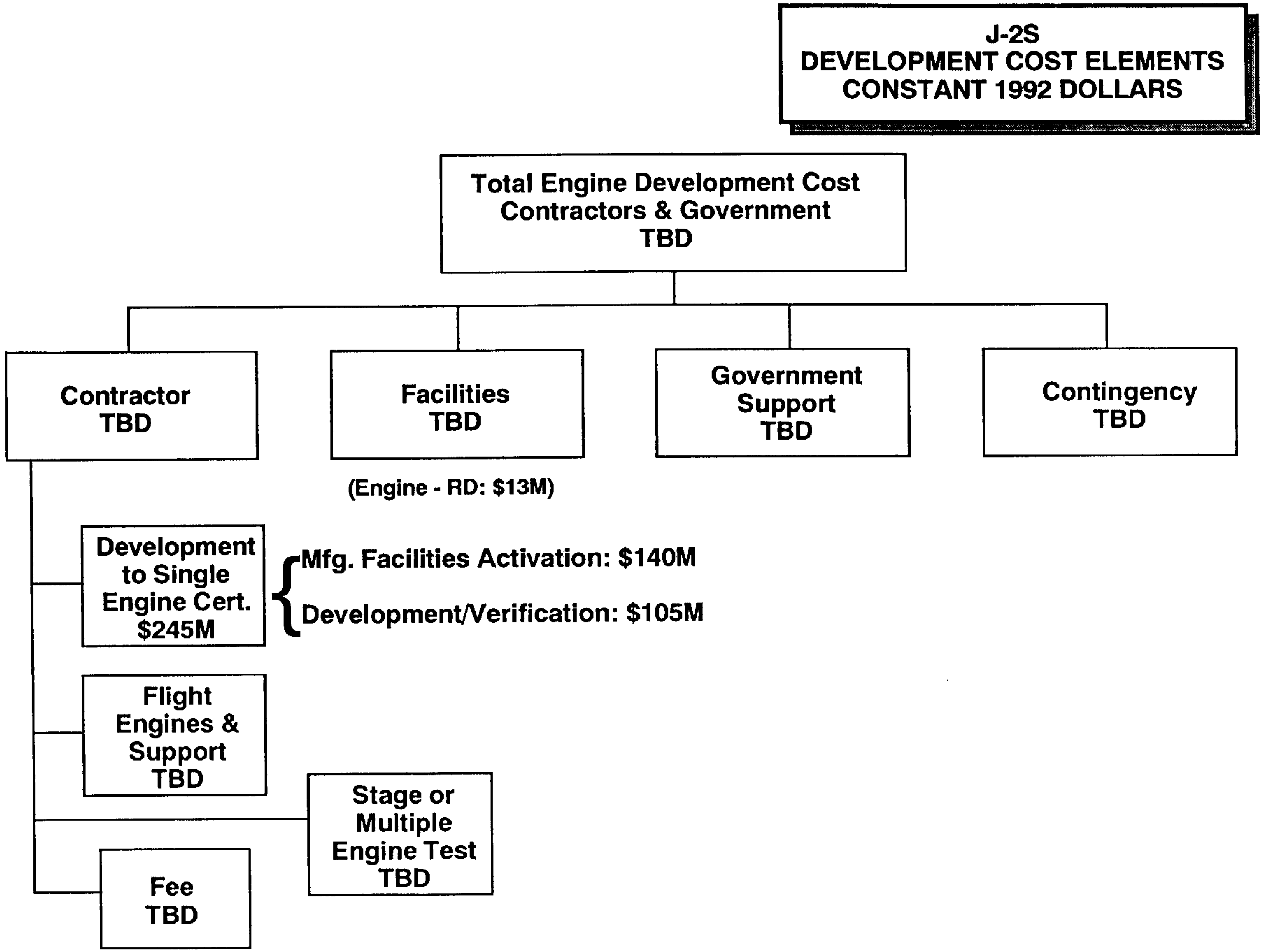


Figure 19. J-2S Restart Program Cost Elements

TA3–0418Fig

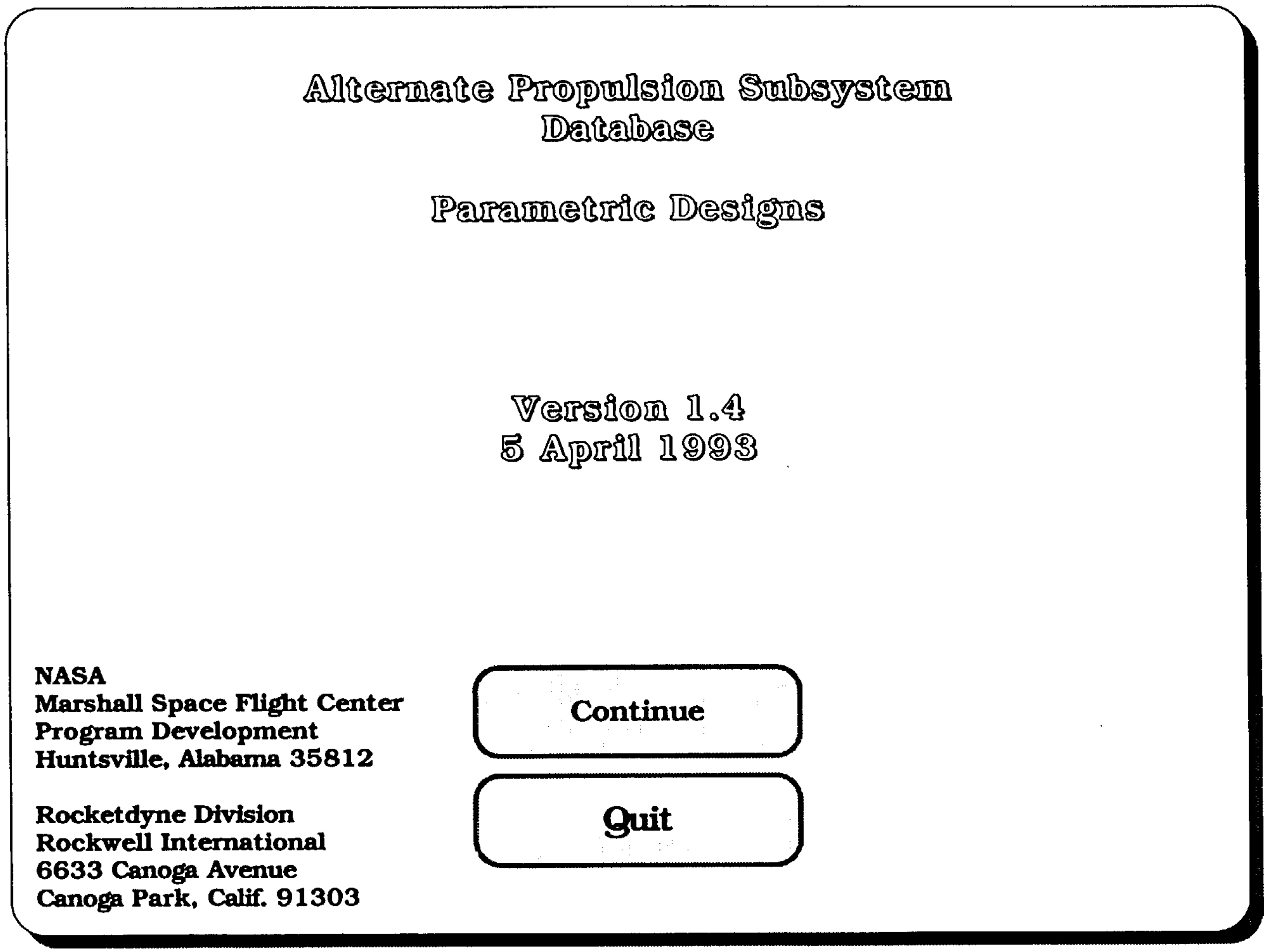


Figure 20. Parametric Database Opening Screen

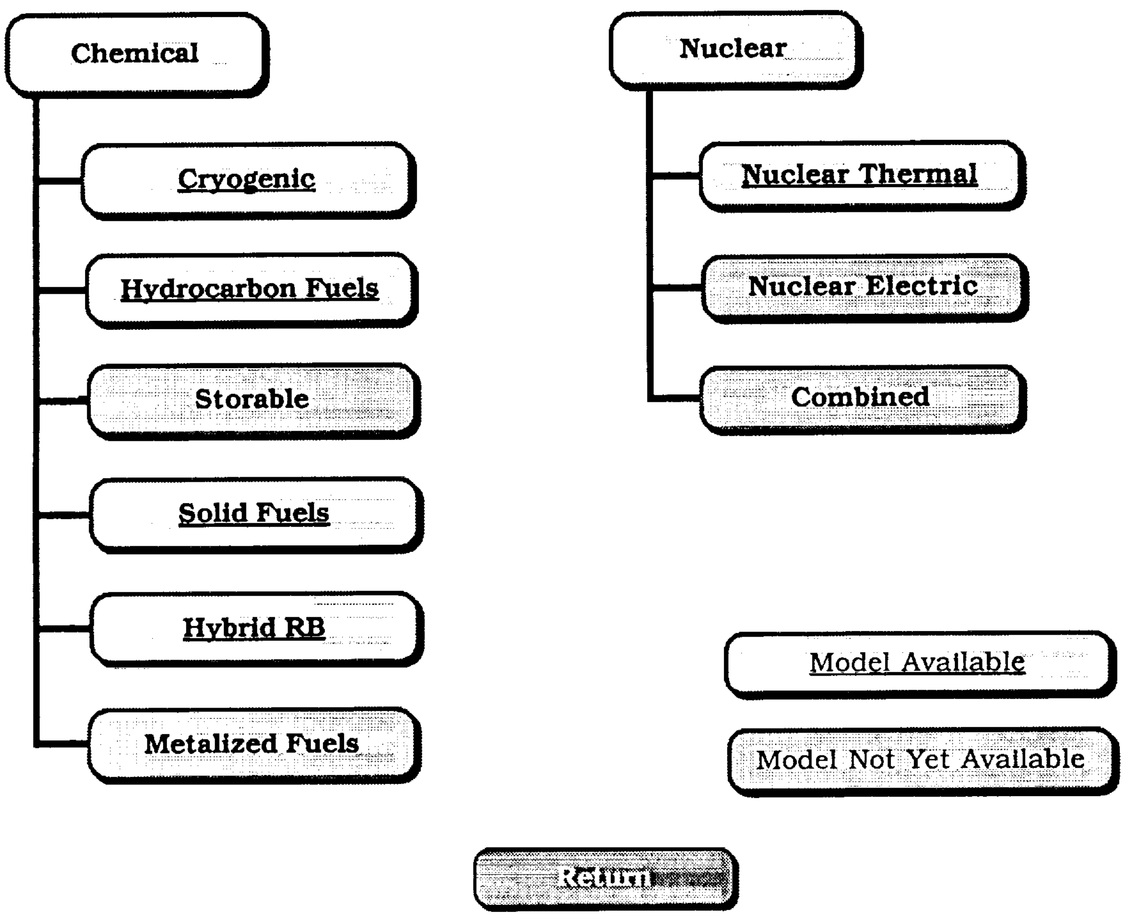


Figure 21. Main Navigation Screen

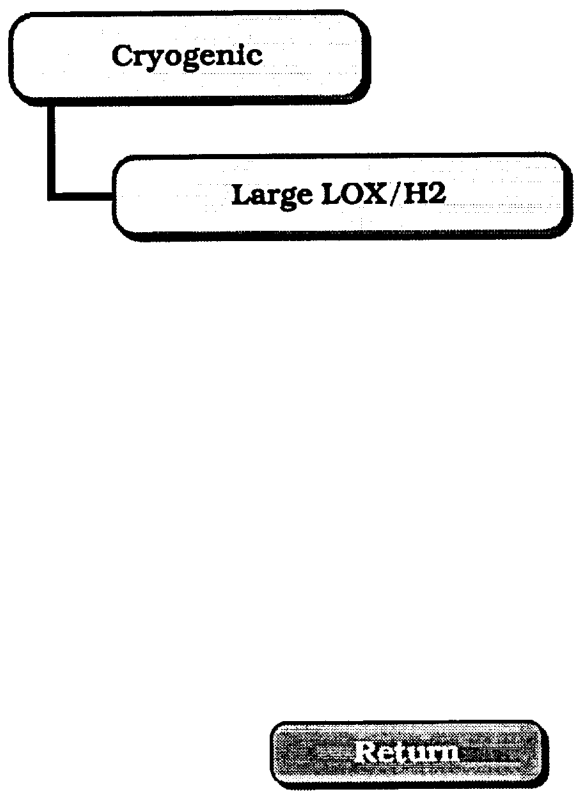


Figure 22. Current Cryogenic Models

Return | English Units | Print (Report) | Print (Briefing)

Calculate – LOX/H2 | Page Down | Graphs

Liquid Engines	LOX/H2	26 January 1993
Independent Terms	**Value**	**Valid Range**
Major Variables		
Vacuum Thrust, klbf	512.845	100 to 2,000
Chamber Pressure, psia	3,277.0	1,000 to 5,000
Mixture Ratio, O/F	6.011	4 to 8
Maximum Area Ratio	77.0	10 to 400
Parameters		
Area Ratio of Nozzle Attachment	5.0	
Nozzle Percent Length, %	80.0	70 to 140
Gimbal Angle, degrees	11.0	0 to 15
C* Efficiency	0.98450	0.85 to 0.999
Fuel Inlet Enthalpy, kcal/mole	-1.270	-2.154 to 1.856

Performance	Value
Vacuum Thrust, klbf	512.845
Vacuum Isp, sec-lbf/lbm	452.983
SL Thrust, klbf	418.772
SL Isp, sec-lbf/lbm	369.891
ODE C-Star, ft/sec	7,753.620
L-Star, in	30.399
ODE Isp, sec-lbf/lbm	468.923
Energy Release Efficiency	0.984
Kinetic Efficiency	1.000
Divergence Efficiency	0.993
Boundary Layer Efficiency	0.989
Engine Efficiency	0.967

Dimensions	Value
Throat Diameter, in	10.3
Throat Area, in^2	83.2
Chamber Length, in	12.3
Nozzle Exit Diameter, in	90.3
Engine Diameter, in	96.0
Nozzle Length, in	119.5
Engine Length, in	168.0

Weights, lbm	Value
Turbomachinery	1,725.0
Preburners	229.0
PB Hot Gas Manifold	558.0
Thrust Chamber	859.0
Nozzle	1,250.0
Gimbal Bearing	105.0
Valves and Controls	722.0
Controller and Mount	85.0
POGO System	94.0
Propellant Ducts	867.7
Pressurization System	89.0
Other Engine Systems	228.0
Total Dry Weight	6,811.7

Return | Page Up

Figure 23. Input/Output Table for Liquid Models

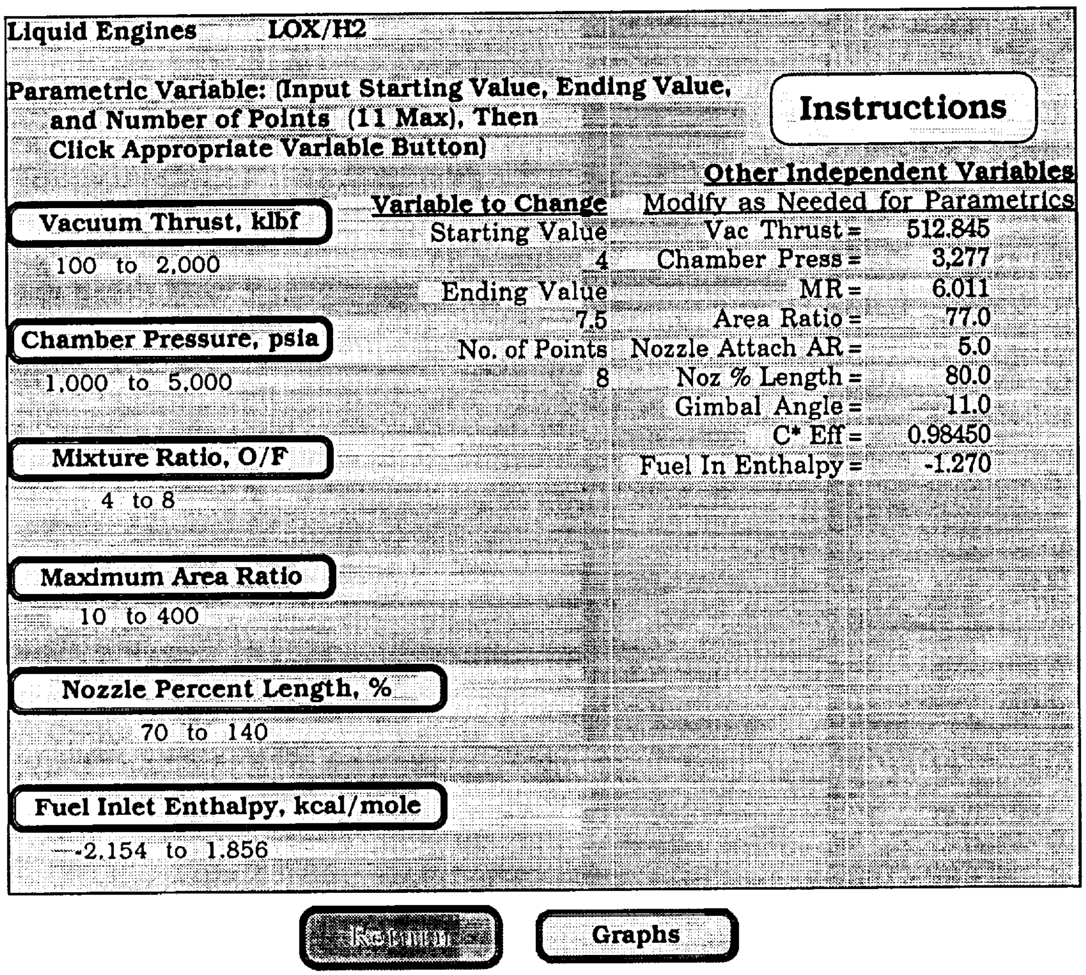


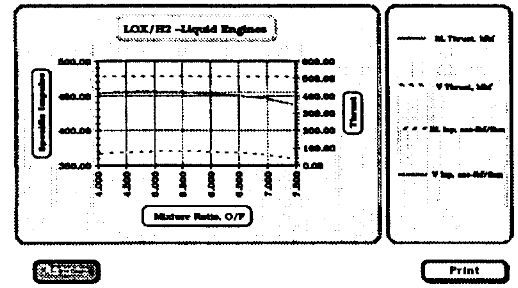
Figure 24. Parametric Data Generation Screen – Liquid Engines

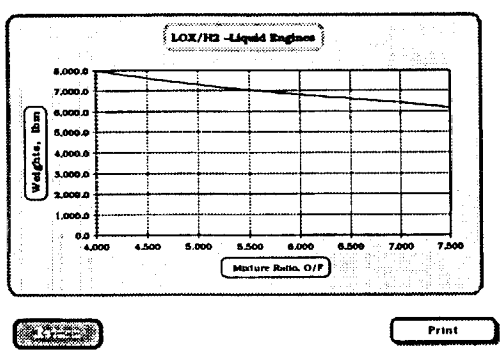
Weight | Lengths | Performance | Print

Return | Page Down | Use the Yellow Buttons to See the Graphs

Independent Variables	LOX/H2 -Liquid Engines								
Vacuum Thrust, lbf	-	512.845	512.845	512.845	512.845	512.845	512.845	512.845	512.845
Chamber Press, psia	-	3.277	3.277	3.277	3.277	3.277	3.277	3.277	3.277
Mixture Ratio, O/F	-	4.000	4.500	5.000	5.500	6.000	6.500	7.000	7.500
Max Area Ratio	-	77.0	77.0	77.0	77.0	77.0	77.0	77.0	77.0
Noz Attach AR	-	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0
Nozzle % Length	-	80.0	80.0	80.0	80.0	80.0	80.0	80.0	80.0
Gimbal Ang, deg	-	11.0	11.0	11.0	11.0	11.0	11.0	11.0	11.0
C* Eff	-	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450
Fuel H, kcal/mole	-	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270
Weights, lbm									
Turbomachinery	-	2,092.8	1,970.6	1,861.4	1,790.6	1,728.5	1,663.5	1,606.9	1,515.6
Preburners	-	363.3	329.4	288.0	255.5	229.5	209.3	190.7	164.7
PB Hot Gas Man	-	660.7	627.4	600.1	577.5	558.4	541.6	527.0	506.3
Thrust Chamber	-	988.8	945.9	911.3	883.3	859.5	838.8	820.8	795.0
Nozzle	-	1,296.1	1,286.1	1,272.7	1,261.3	1,250.2	1,239.4	1,229.1	1,214.8
Gimbal Bearing	-	105.0	105.0	105.0	105.0	105.0	105.0	105.0	105.0
Valves & Cont	-	780.6	768.1	751.3	736.8	722.3	709.8	696.7	681.7
Cont & Mount	-	85.0	85.0	85.0	85.0	85.0	85.0	85.0	85.0
POGO System	-	94.6	94.7	94.6	94.3	94.0	93.6	93.1	92.5
Prop Ducts	-	1,110.5	1,037.0	971.1	915.6	866.7	827.8	791.6	736.8
Pres Sys	-	93.6	92.3	91.1	90.0	89.0	88.0	87.1	85.7
Other Engine Sys	-	275.7	261.6	248.7	237.8	228.2	219.7	212.3	200.1
Total Dry Weight	-	7,964.7	7,611.1	7,300.6	7,039.8	6,816.3	6,619.4	6,442.2	6,190.6
Dimensions									
Throat Dia, in	-	10.5	10.5	10.4	10.3	10.3	10.2	10.2	10.1
Throat Area, in^2	-	86.6	85.6	84.9	84.1	83.2	82.4	81.7	80.6
Chamber Len, in	-	12.5	12.4	12.4	12.3	12.3	12.3	12.2	12.1
Noz Exit Dia, in	-	92.2	91.7	91.3	90.8	90.3	89.9	89.5	88.9
Engine Dia, in	-	96.0	96.0	96.0	96.0	96.0	96.0	96.0	96.0
Noz Length, in	-	122.0	121.3	120.7	120.1	119.5	118.9	118.4	117.6
Engine Length, in	-	171.4	170.5	169.6	168.8	168.0	167.2	166.5	165.4
Performance									
SL Thrust, klbf	-	414.76	415.82	416.87	417.83	418.75	419.66	420.53	421.75
V Thrust, klbf	-	512.84	512.84	512.84	512.84	512.84	512.84	512.84	512.84
SL Isp, sec-lbf/lbm	-	366.75	369.26	370.38	370.47	369.61	368.28	365.84	356.88
V Isp, sec-lbf/lbm	-	453.50	456.42	456.43	454.71	453.08	450.06	446.18	436.36
ODE C*, ft/sec	-	8.095	8.039	7.966	7.861	7.758	7.631	7.495	7.233
L-Star, in	-	30.7	30.6	30.6	30.5	30.4	30.3	30.3	30.2
ODE Isp	-	469.44	471.42	471.63	470.08	468.57	465.93	461.97	453.24
Energy Rel Eff	-	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450
Kinetic Eff	-	0.99999	0.99998	0.99999	0.99999	0.99994	0.99986	0.99968	0.99669
Divergence Eff	-	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283
BL Eff	-	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904
Engine Eff	-	0.96672	0.96672	0.96672	0.96672	0.96666	0.96658	0.96643	0.96344

Return | Page Up





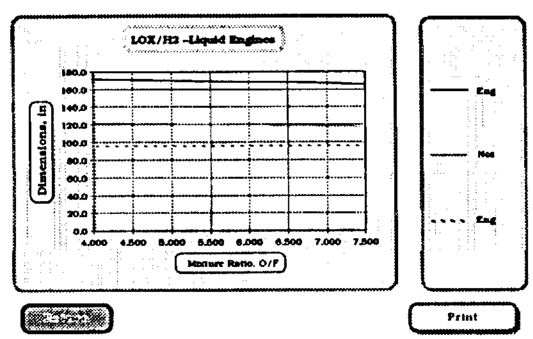


Figure 25. Parametric Data Available – Liquid Engines

Independent Variables	LOX/H2 –Liquid Engines							
Vacuum Thrust, lbf =	512.845	512.845	512.845	512.845	512.845	512.845	512.845	512.845
Chamber Press, psi =	3,277	3,277	3,277	3,277	3,277	3,277	3,277	3,277
Mixture Ratio, O/F =	4.000	4.500	5.000	5.500	6.000	6.500	7.000	7.500
Max Area Ratio =	77.0	77.0	77.0	77.0	77.0	77.0	77.0	77.0
Noz Attach AR =	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0
Nozzle % Length =	80.0	80.0	80.0	80.0	80.0	80.0	80.0	80.0
Gimbal Ang, deg =	11.0	11.0	11.0	11.0	11.0	11.0	11.0	11.0
C* Eff =	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450
Fuel H, kcal/mole =	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270	-1.270
Weights, lbm								
Turbomachinery =	2,092.9	1,978.6	1,881.6	1,798.6	1,726.5	1,662.5	1,605.9	1,515.6
Preburners =	383.3	329.4	288.0	255.5	229.5	208.3	190.7	164.7
PB Hot Gas Man =	660.7	627.4	600.1	577.5	558.4	541.6	527.0	505.3
Thrust Chamber =	988.9	945.9	911.3	883.2	859.5	838.8	820.9	795.0
Nozzle =	1,298.1	1,285.1	1,272.7	1,261.2	1,250.2	1,239.4	1,229.1	1,214.5
Gimbal Bearing =	105.0	105.0	105.0	105.0	105.0	105.0	105.0	105.0
Valves & Cont =	789.6	769.1	751.3	735.9	722.3	709.9	698.7	681.7
Cont & Mount =	85.0	85.0	85.0	85.0	85.0	85.0	85.0	85.0
POGO System =	94.6	94.7	94.6	94.3	94.0	93.6	93.1	92.5
Prop Ducts =	1,116.5	1,037.0	971.1	915.9	868.7	827.5	791.6	735.5
Pres Sys =	93.6	92.3	91.1	90.0	89.0	88.0	87.1	85.7
Other Engine Sys =	276.7	261.6	248.7	237.8	228.2	219.7	212.2	200.1
=								
=								
=								
=								
=								
Total Dry Weight =	7,984.7	7,611.1	7,300.5	7,039.9	6,816.3	6,619.4	6,446.2	6,180.6
Dimensions								
Throat Dia, in =	10.5	10.5	10.4	10.3	10.3	10.2	10.2	10.1
Throat Area, in^2 =	86.8	85.8	84.9	84.1	83.2	82.4	81.7	80.6
Chamber Len, in =	12.5	12.4	12.4	12.3	12.3	12.2	12.2	12.1
Noz Exit Dia, in =	92.2	91.7	91.2	90.8	90.3	89.9	89.5	88.9
Engine Dia, in =	96.0	96.0	96.0	96.0	96.0	96.0	96.0	96.0
Noz Length, in =	122.0	121.3	120.7	120.1	119.5	118.9	118.4	117.6
Engine Length, in =	171.4	170.5	169.6	168.8	168.0	167.2	166.5	165.4
Performance								
SL Thrust, klbf =	414.74	415.82	416.87	417.83	418.75	419.66	420.53	421.75
V Thrust, klbf =	512.84	512.84	512.84	512.84	512.84	512.84	512.84	512.84
SL Isp, sec-lbf/lbm =	366.75	369.26	370.35	370.47	369.91	368.28	365.84	358.85
V Isp, sec-lbf/lbm =	453.50	455.42	455.62	454.71	453.03	450.06	446.15	436.36
ODE C*, ft/sec =	8,095	8,039	7,956	7,861	7,756	7,631	7,495	7,233
L-Star, in =	30.7	30.6	30.6	30.5	30.4	30.3	30.3	30.2
ODE Isp =	469.44	471.42	471.62	470.69	468.97	465.93	461.97	453.24
Energy Rel Eff =	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450	0.98450
Kinetic Eff =	0.99999	0.99999	0.99999	0.99999	0.99994	0.99986	0.99968	0.99660
Divergence Eff =	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283	0.99283
BL Eff =	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904	0.98904
Engine Eff =	0.96672	0.96672	0.96672	0.96672	0.96666	0.96658	0.96642	0.96344

Figure 26. Printed Version of Parametric Results Chart - Liquid Engines

Alternate Propulsion Subsystem Concepts
Database
Version 1.3

5 April 1993

NASA
Marshall Space Flight Center
Program Development
Huntsville, Alabama 35812

Continue

Rocketdyne Division
Rockwell International
6633 Canoga Avenue
Canoga Park CA 91303

Figure 27. Propulsion System Database Opening Screen

May 14, 1993

Propulsion System Menu

Please click Icon for selected propulsion system

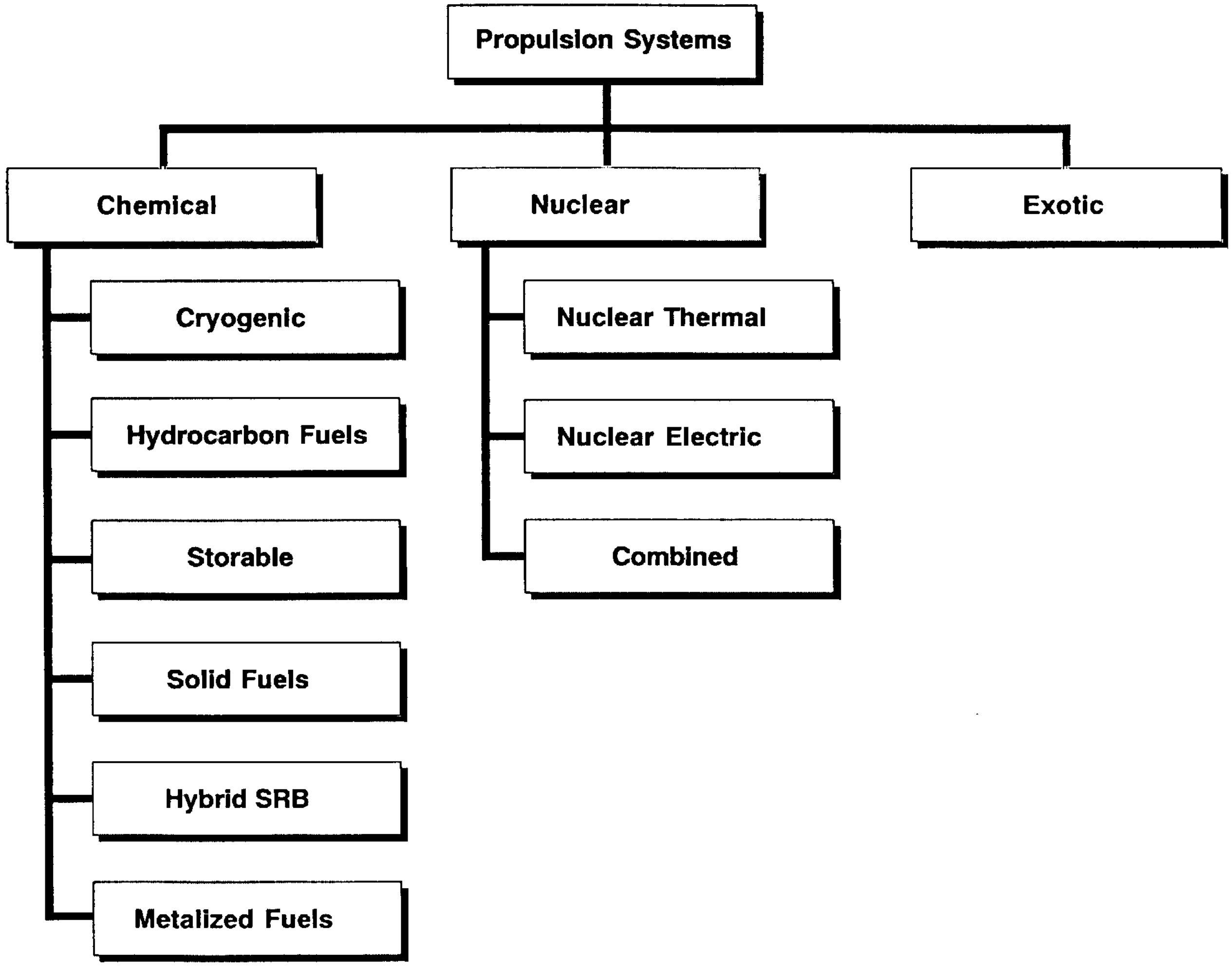


Figure 28. Main Menu of Propulsion Types

Print More Data

Summary of Propulsion Systems

Reports Data Entry

	Engine Name	Acronym	Engine Class
1	Space Transportation Main Engine	STME	Cryogenic Liquid
2	F-1	F-1	Hydrocarbon Liquid
3	F-1A	F-1A	Hydrocarbon Liquid
4	J-2	J-2	Cryogenic Liquid
5	Simplified, High Performance J-2	J-2S	Cryogenic Liquid
6	Space Shuttle Main Engine	SSME	Cryogenic Liquid
7	RD-170	RD-170 (Russian Designation 11D521)	Hydrocarbon Liquid
	Integrated Modular Engine	IME	Cryogenic Liquid
9	Space Shuttle Redesigned Solid Rocket Motor	RSRM	Solid Fuel

Figure 29. Propulsion Systems Currently Available

Liquids

May 14, 1993

Reports

Summary

Engine Reports

- Background Data
- Propulsion System Basic Information
- Engine Performance Report #1
- Engine Performance Report #2
- Start-Up/Shutdown Sequence Report
- Start-Up/Shutdown Profile #1
- Start-Up/Shutdown Profile #2
- Interface Report
- Engine Technology Development
- Advanced Development Plan
- Engine Picture/Basic Data
- Engine Drawing
- Engine Balance

Engine Briefing Charts

Propulsion Element Data

- Chart #1
- Chart #2
- Chart #3
- Chart #4
- Chart #5
- Chart #6

- Background Data
- Startup Sequence
- Shutdown Sequence
- Interface Chart
- Engine Technology Development
- Advanced Development Plan
- Thrust Startup/Shutdown Profile
- Specific Impulse Startup/Shutdown Profile
- Mixture Ratio Startup/Shutdown Profile
- Mass Flow Startup/Shutdown Profile
- Engine Picture/Basic Data
- Engine Drawing
- Engine Balance

Figure 30. Reports Available for Each Propulsion System

May 14, 1993

Engine Performance 1

Print | Reports

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

Propellants

Oxidizer	Liquid Oxygen	
Fuel	Liquid Hydrogen	
Mixture Ratio – Engine/Thrust Chamber	6.000	6.993

Nominal Chamber Pressure	2,250
Expansion Ratio	45.00
Engine Design Life (Flights)	1

Engine Restarts

Design	0
Demonstrated	

Engine Thrust Data

	Sea Level	Vacuum
Nominal	552,980	650,000
Maximum		
Minimum	357,980	455,000

Thrust data in units of lbf

Engine Starts

Design	11
Demonstrated	

Engine Reliability, sec

Design	5,500
Demonstrated	

Throttle Ratio, Percent

	Sea Level	Vacuum
Maximum		
Minimum	64.70	70.00

Specific Impulse Data

	Sea Level	Vacuum
@Nominal Thrust	364.54	428.50
@Maximum Thrust		
@Minimum Thrust	336.74	428.00

Specific Impulse data in units of seconds

Nozzle Data

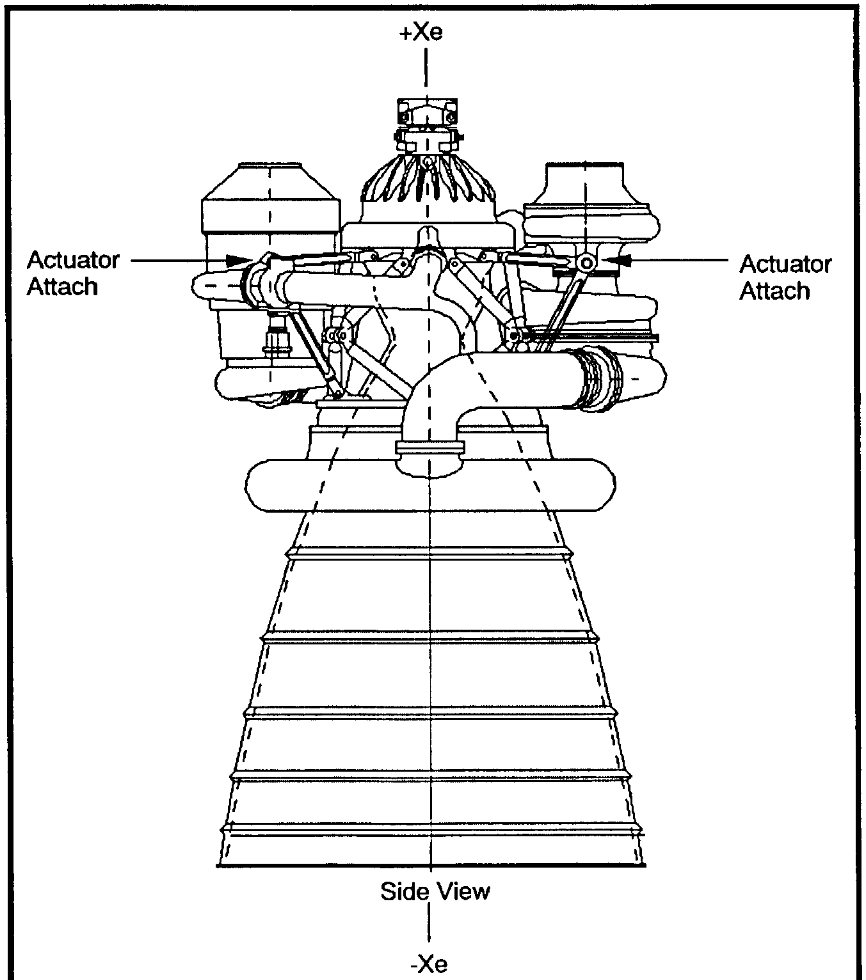
Type	Bell
Length (in)	116.00
Diameter (in)	91.67
Throat Area (sq. in)	146.61
Exit Area (sq. in)	6,597.45
Expansion Ratio	45.00

Figure 31. Typical Report Page Layout

Figure 32.

Output for Space Transportation Main Engine (STME) Propulsion System

STME Propulsion System

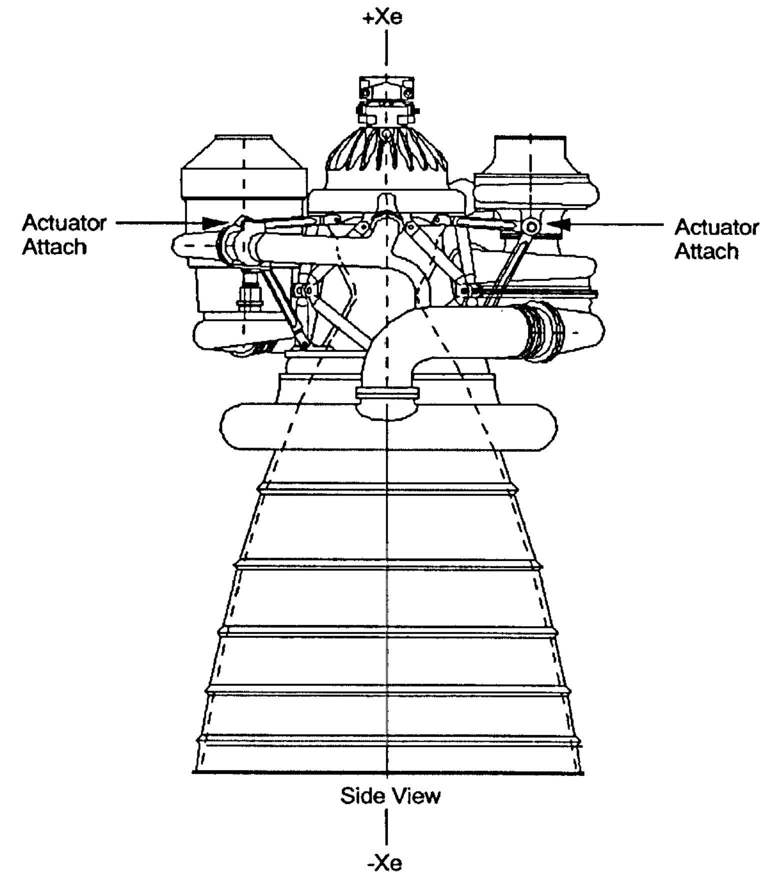


- Nominal Thrust (lbf)
  - Sea Level — 552,980
  - Vacuum — 650,000
- Specific Impulse (sec)
  - Sea Level — 364.5
  - Vacuum — 428.5
- Chamber Pressure (psia) (Nozzle Stagnation) — 2,250
- Engine Mixture Ratio — 6.000
- Expansion Ratio — 45.00
- Length (in) — 161.00
- Weight (lbm) — 9,100

5/14/93

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical



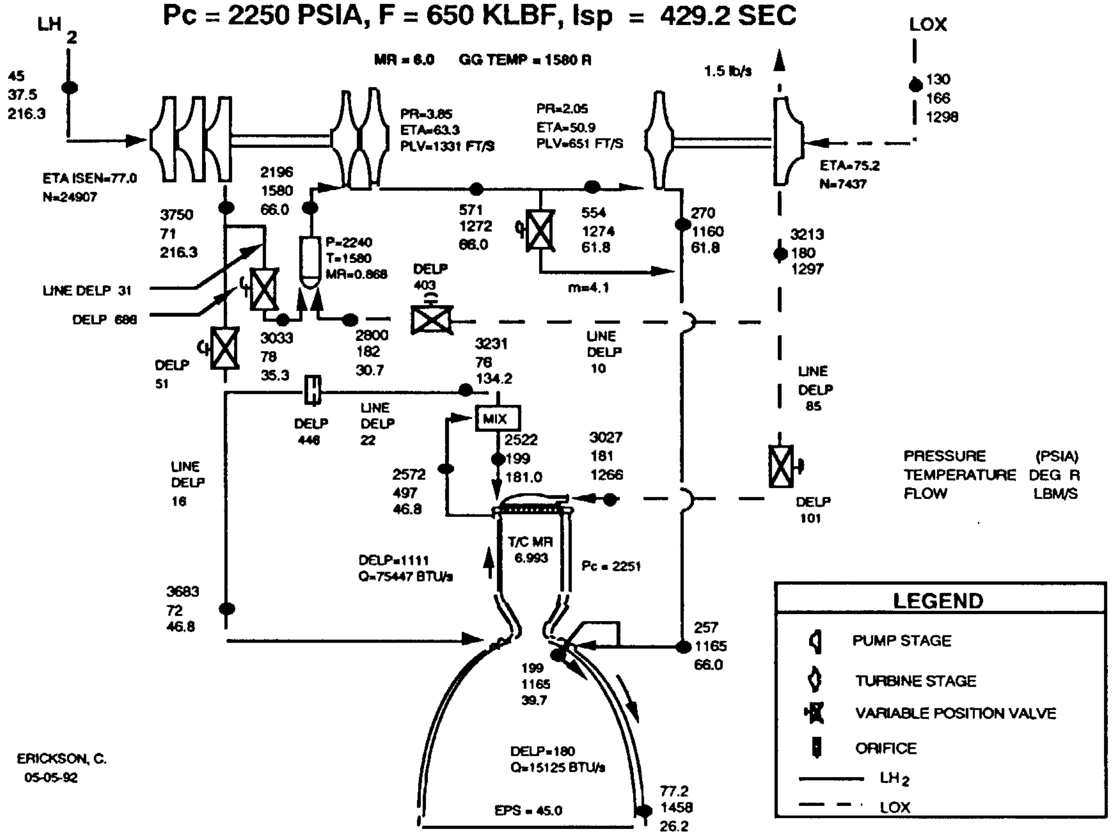
91

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

STEP - REV. 26b @ RPL INLET

Pc = 2250 PSIA, F = 650 KLBF, Isp = 429.2 SEC



Background Information

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

Background

The STME was designed to support propulsion requirements of the National Launch System (NLS). The NLS concept provides a lift capacity for a family of launch vehicles with a wide range of payload sizes (approximately 20,000 lbs and above) and missions. NLS family members may consist entirely of liquid propulsion units or combinations of liquid units and solid rocket motors.

The STME is capable of operating in either a NLS booster or core propulsion application. In either mode, the STME starts prior to vehicle liftoff. In the booster mode, the operation of some STME's will be terminated and detached from the vehicle with other elements while other STME's continue to operate.

In the core mode, the STME will continue to operate after booster (solid or liquid) separation until orbital (or near orbital) conditions are reached.

The STME is a pump fed liquid oxygen and liquid hydrogen engine that has been designed for high reliability and low cost. It employs a gas generator power cycle to drive separate LO2 and LH2 turbopump assemblies. Gas generator propellants are tapped-off the engine propellant system and burned to provide fuel rich gas to drive the turbines. Turbine exhaust gas is used to cool the engine nozzle extension. The engine is capable of operating at two discrete thrust levels, 100% and 70%. Engine start is accomplished by use of vehicle propellant tank head pressures. No helium spin start or solid start cartridge is required. The engine provides oxygen and hydrogen gases for propellant tank pressurization.

Comments

References

Source: STME Technical Information Document , 6 Jan 1993; ICD, Working Draft, Attachment J-3, 18 Sept 1992; Draft Contract End Item Specification, Phase C/D, Revision 10, Attachment J-2, 26 May 1992

Date: Entered as of 31 March 1993

Entered by: Dan Levack

Propulsion System General Data

May 14, 1993

Creation Date	Modification Date	Record Number
3/18/93	3/31/93	1

Engine Name	Space Transportation Main Engine	
Class of Engine	Cryogenic Liquid	Chemical
Propulsion Type	Thermodynamic Expansion of Hot Gas	
Acronym	STME	
Application	Booster Engine	
Manufacturer	Consortium (Aerojet, Pratt & Whitney, Rocketdyne)	
Program Status	Detailed Study	
Manrated		
IOC/Date Studied (Month/Year)	12/1992	
Mixture Ratio – Engine/ Thrust Chamber	6.000	6.993

Propellants

Oxidizer	Liquid Oxygen
Fuel	Liquid Hydrogen

Engine Design Life (Flights)	1
Restart Capability	No
Engine Cycle	Gas Generator
Nominal Chamber Pressure	2,250
Expansion Ratio	45.00
TVC Method	Gimbal

Dimensions

Maximum Length (inches)	161.00
Maximum Width (inches)	101.22
Engine Mass (lbm)	9,100.00

Engine Thrust Data, lbf

	Sea Level	Vacuum
Nominal	552,980	650,000
Maximum		
Minimum	357,980	455,000

May 14, 1993

Engine Performance 1

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

Propellants

Oxidizer	Liquid Oxygen	
Fuel	Liquid Hydrogen	
Mixture Ratio – Engine/Thrust Chamber	6.000	6.993

Nominal Chamber Pressure	2,250
Expansion Ratio	45.00
Engine Design Life (Flights)	1

Engine Restarts

Design	0
Demonstrated	

Engine Thrust Data

	Sea Level	Vacuum
Nominal	552,980	650,000
Maximum		
Minimum	357,980	455,000

Thrust data in units of lbf

Engine Starts

Design	11
Demonstrated	

Engine Reliability, sec

Design	5,500
Demonstrated	

Throttle Ratio, Percent

	Sea Level	Vacuum
Maximum		
Minimum	64.70	70.00

Specific Impulse Data

	Sea Level	Vacuum
@Nominal Thrust	364.54	428.50
@Maximum Thrust		
@Minimum Thrust	336.74	428.00

Specific Impulse data in units of seconds

Nozzle Data

Type	Bell
Length (in)	116.00
Diameter (in)	91.67
Throat Area (sq. in)	146.61
Exit Area (sq. in)	6,597.45
Expansion Ratio	45.00

May 14, 1993

Engine Performance 2

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid   Chemical

Engine Mass (lbm)

Total Mass w/TVC	9,100.0
Total Mass wo/TVC	

TVC

Method	Gimbal
Mass (lbm)	
Max Gimbal Angle (deg)	11.3
Max Gimbal Rate (deg/s)	4.7

Engine Cycle

Type: Gas Generator

Pressures

Oxidizer Turbopump

Min Pump Inlet	47
Turbine Inlet	554.0

Fuel Turbopump

Min Pump Inlet	32
Turbine Inlet	2,196.0

Pressures in psia

Envelope

Length

Nominal	161
Stowed	
Extended	
Maximum Gimbal	

Diameter

Nozzle Exit	97.0
Maximum	101.2
Maximum Gimbal	

Envelope Dimensions in inches

Engine Component Masses

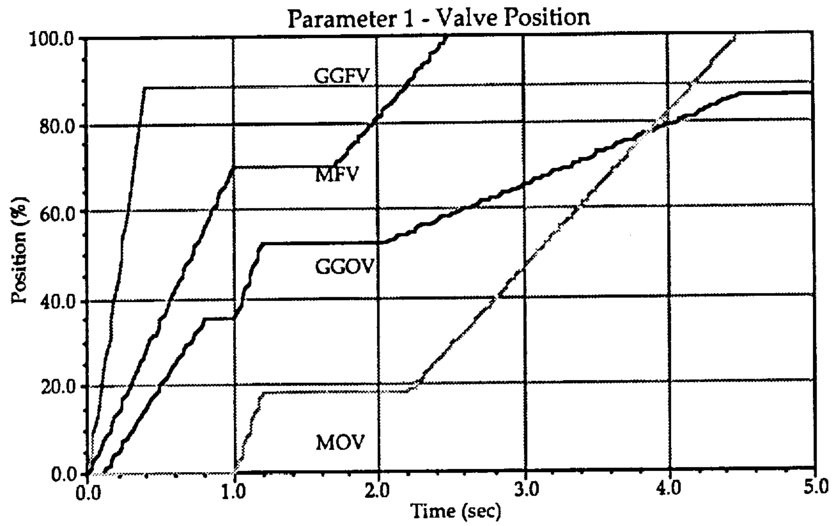
Component	Allocations
Turbomachinery	
Oxygen Turbopump	1570
Fuel Turbopump	1718
Combustion Devices	
Main Injector	1226
Combustion Chamber	1601
Nozzle	1729
Gas Generator	92
Igniter - CC	7
- GG	7
Controls	
Controller	35
Sensors	35
Valves/Actuators	214
Interconnects	17
Pneumatic System	18
Propellant Feed	
Ducts	323
Miscellaneous (System Hardware)	353
Support Devices	
Gimbal System	136
Heat Exchanger	19
Engine Total	9100

Start-Up/Shutdown Sequences

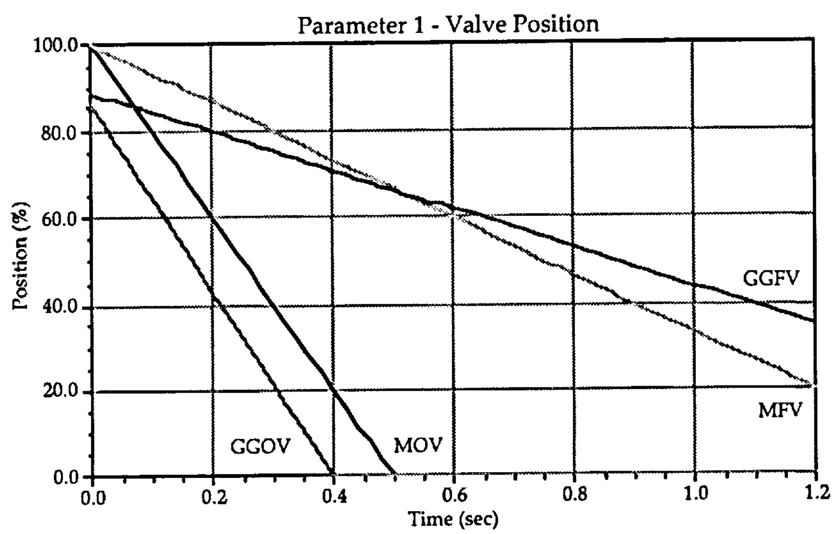
Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

StartUp Sequence



Shutdown Sequence

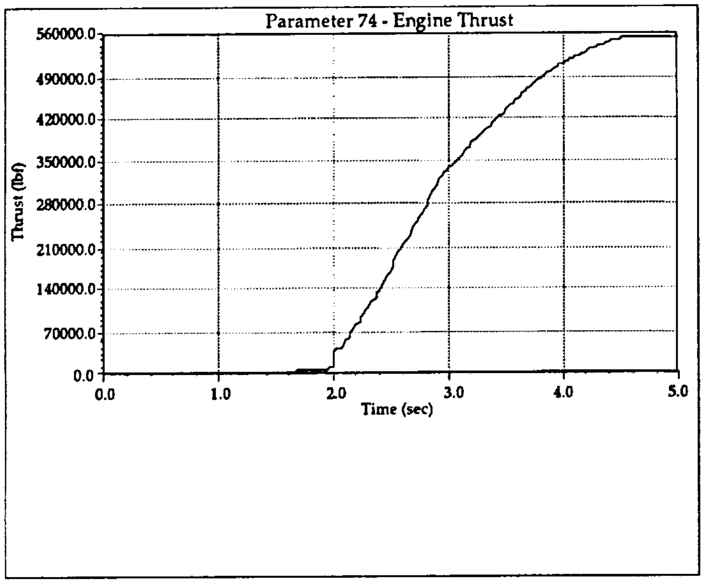


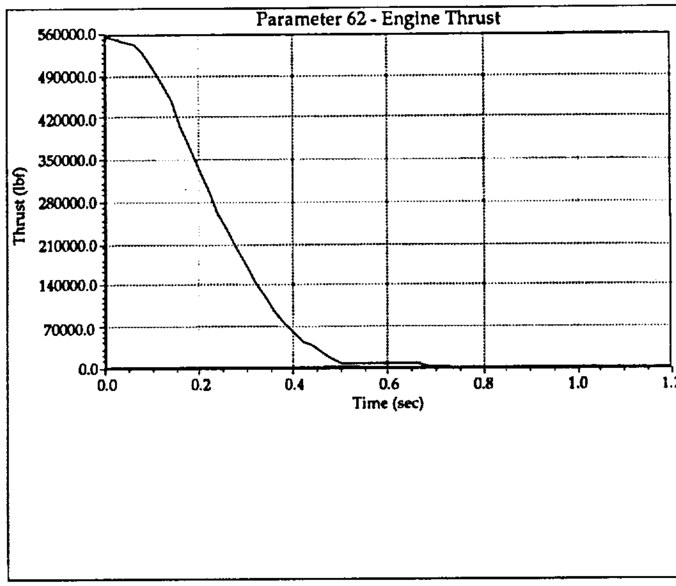
Start-Up/Shutdown Profiles

Engine Name: Space Transportation Main Engine

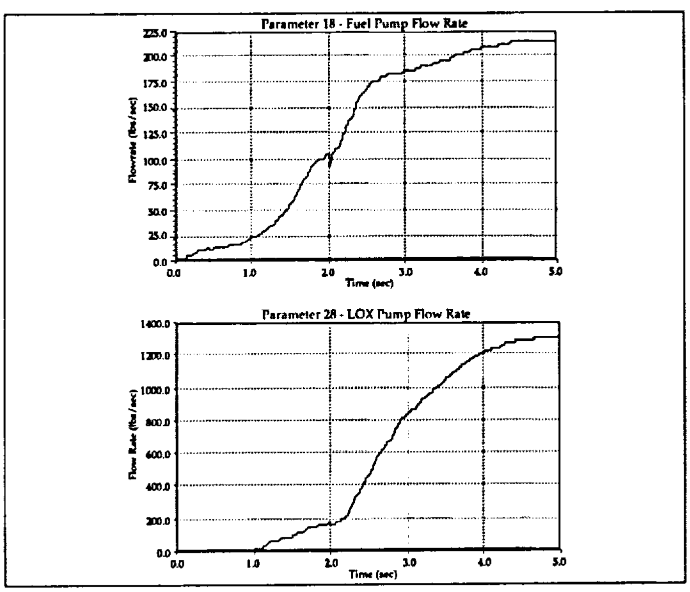
Class of Engine: Cryogenic Liquid Chemical

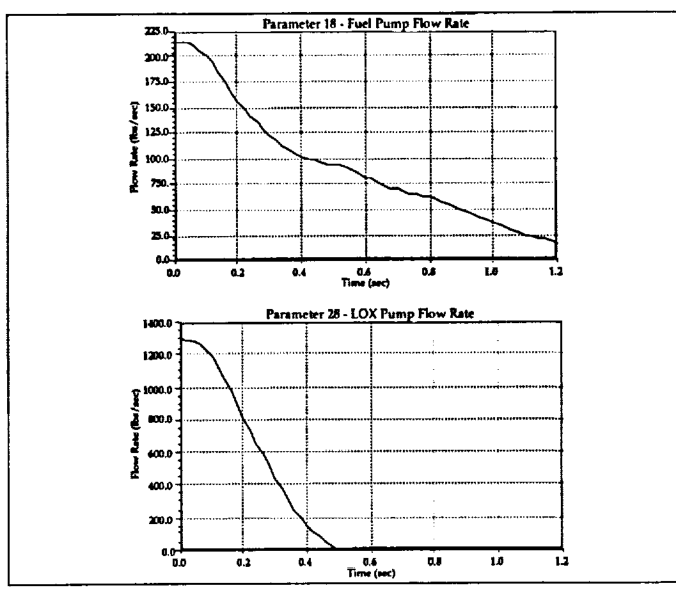
Thrust Profile





Flowrate Profile





May 14, 1993

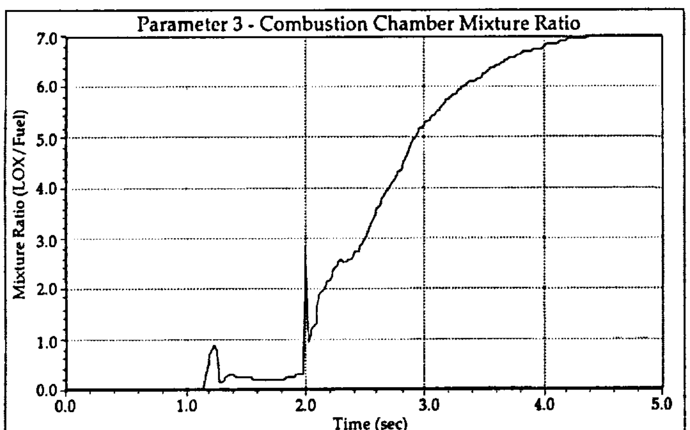
Start-Up/Shutdown Profiles

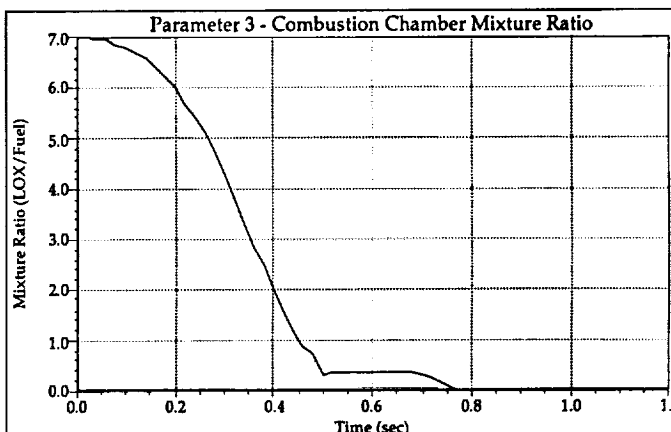
Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid — Chemical

Isp Profile

Mixture Ratio Profile



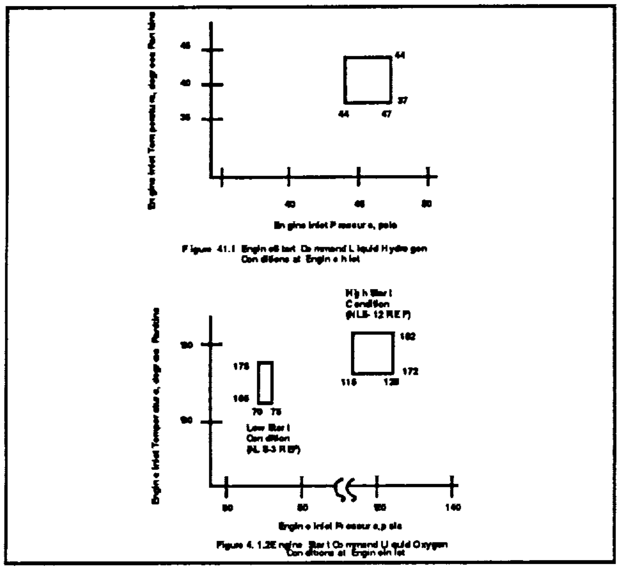


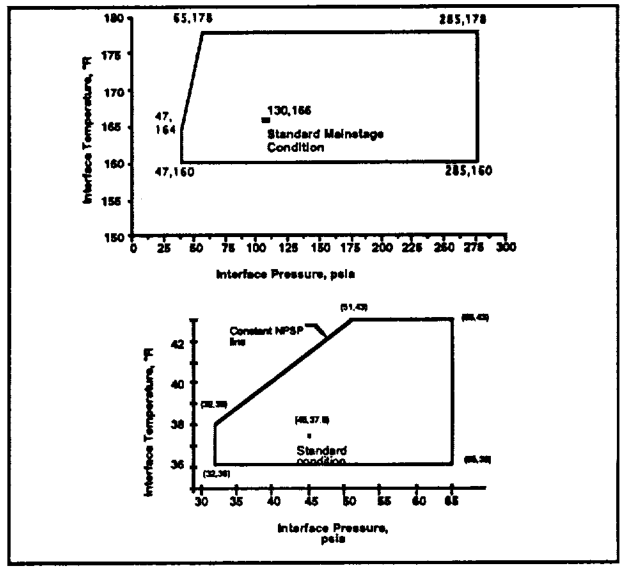
Interfaces

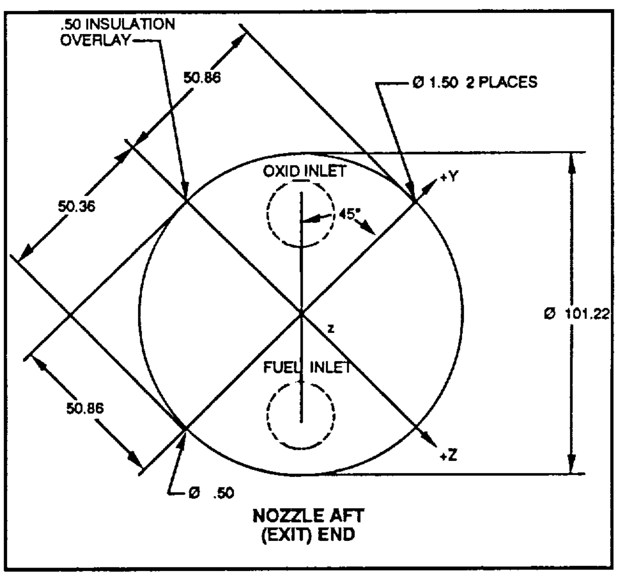
Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

Interfaces







Technology Development

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid Chemical

Technology Development

May 14, 1993

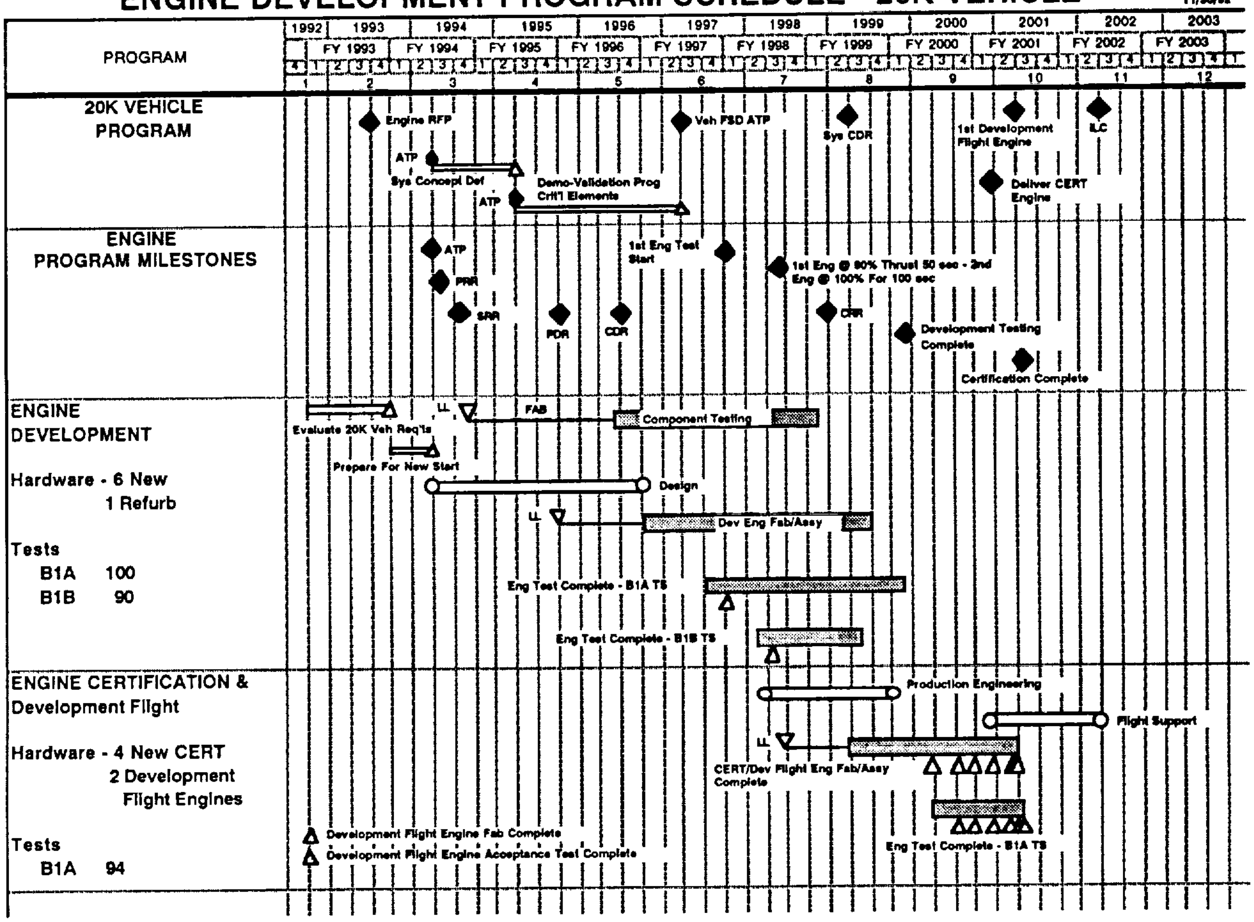
Advanced Development Plan

Engine Name: Space Transportation Main Engine

Class of Engine: Cryogenic Liquid — Chemical

Advanced Development Plan

ENGINE DEVELOPMENT PROGRAM SCHEDULE - 20K VEHICLE



SSME Upper Stage Use

	Thermal	Pressure	Environmental
Altitude Start	• Same as Ground Start	• Lower Pressures • Changed Ratio of LOX/H2 Pressures	• Same as Ground Start
Orbital Start	• Engine Fired • Latent Heat • Solar and Earth Radiation • Heat Redistribution	• Same as Altitude Start	• Engine Fired • Moisture • Flow Path to Vacuum

Figure 33. Impacts of Altitude Start and Orbital Restart on SSME Start Sequence

TA3-0271aFig

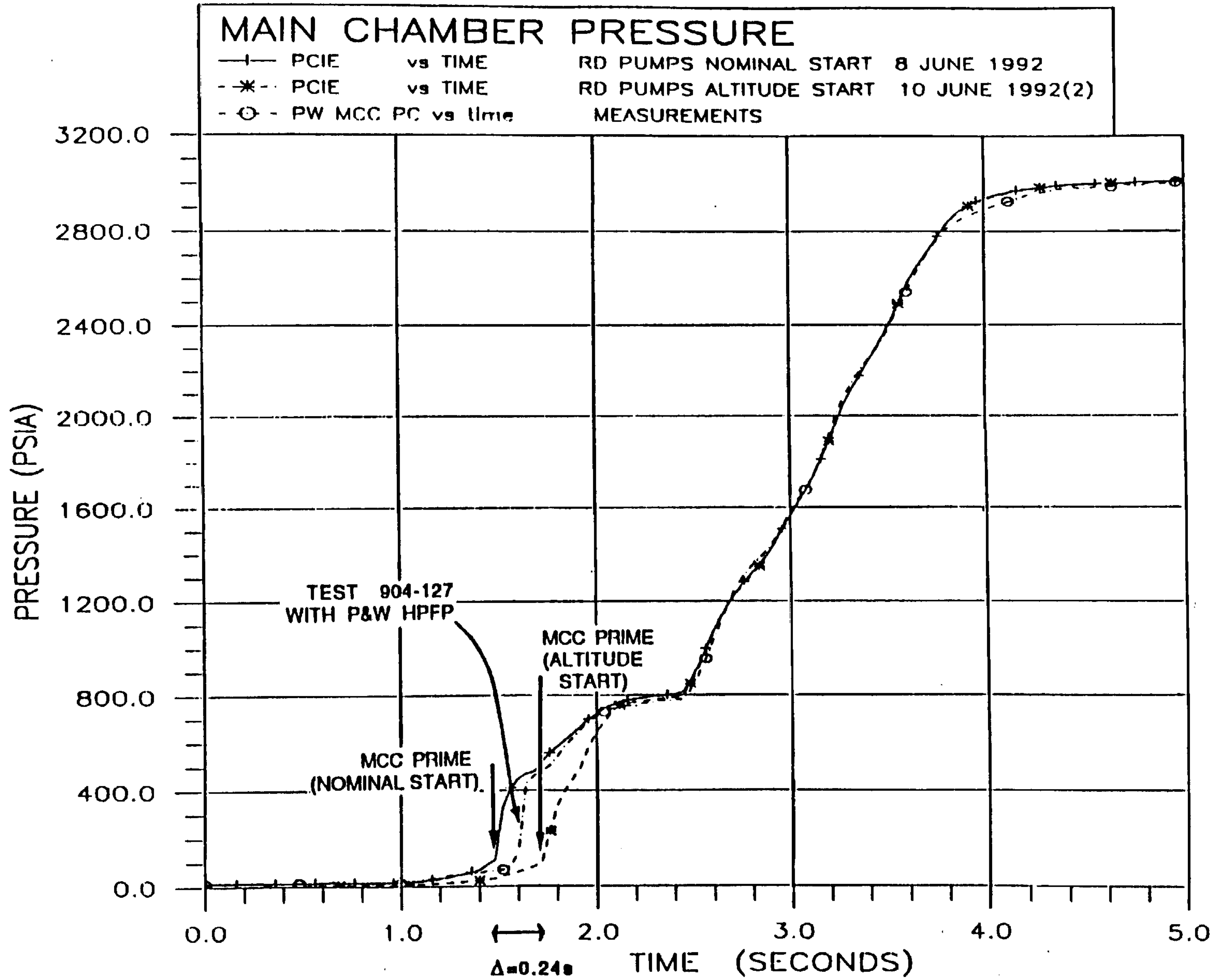


Figure 34. Altitude Start Main Combustion Chamber Pressure

TA3-0132Fig

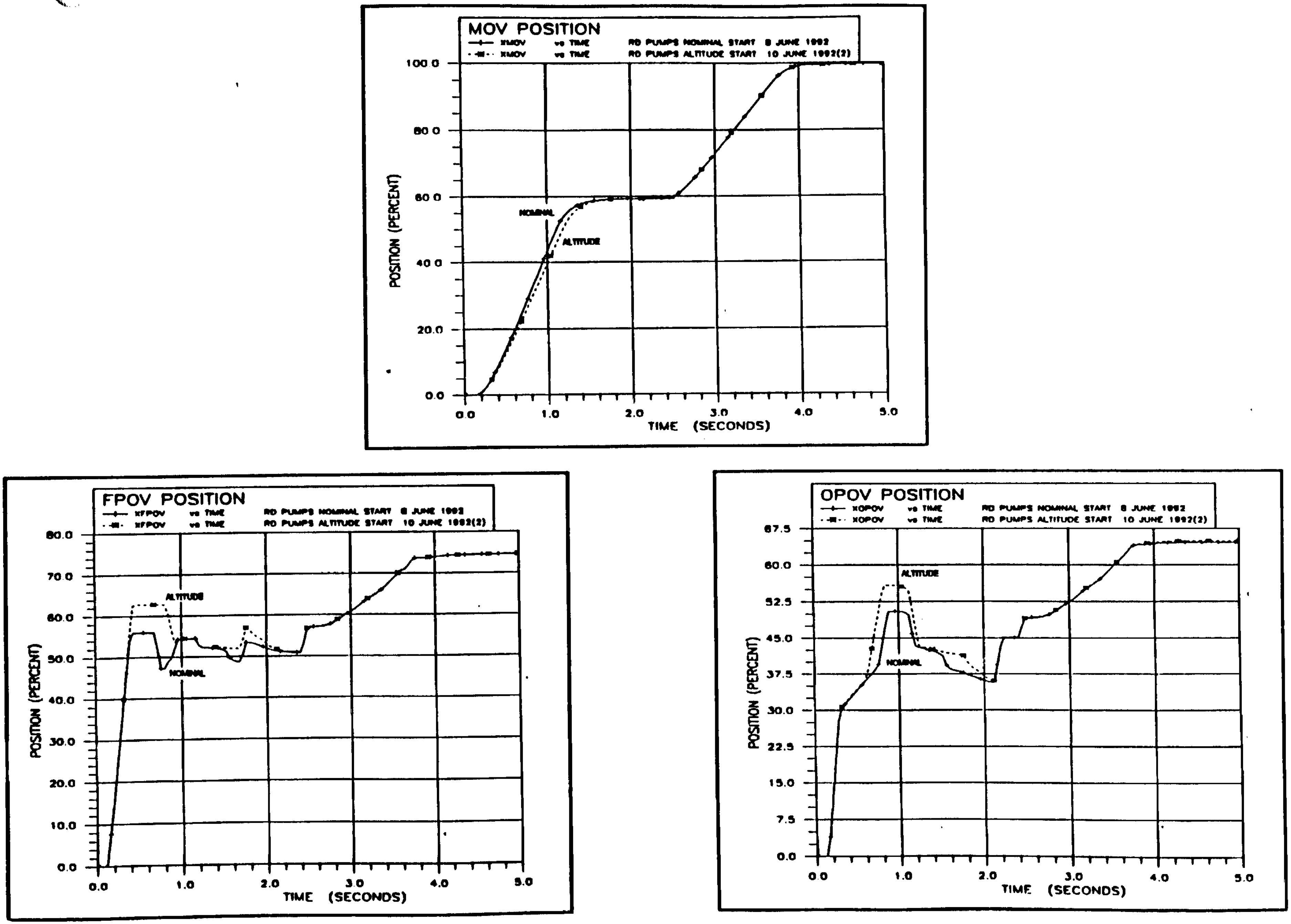


Figure 35. Valve Sequencing

TA3-0130Fig

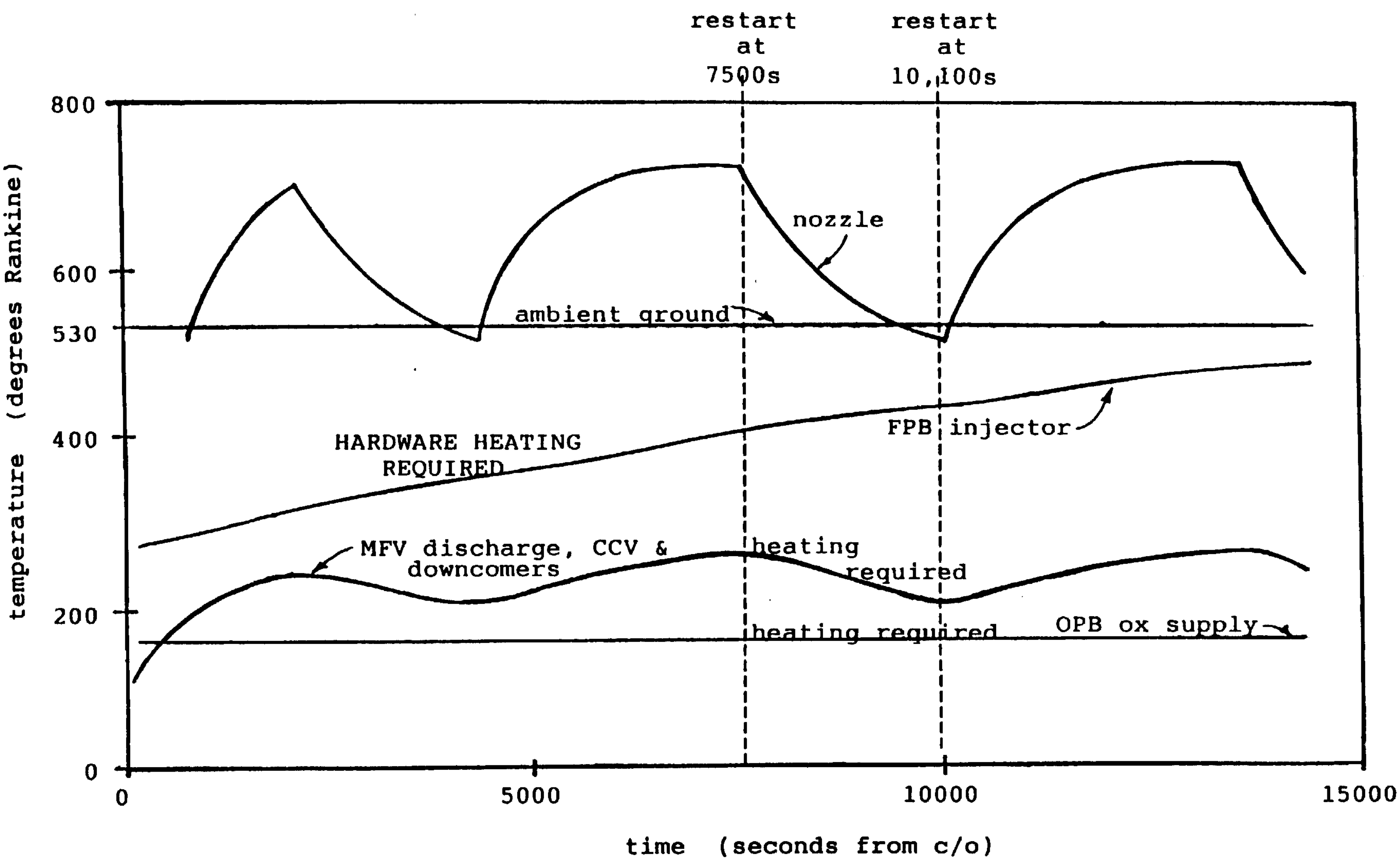


Figure 36. Thermal Analysis Results for Key Components

TA3–0286Fig

SSME Upper Stage Use

Rotating Stage in Sunlight

Case	Recirculation Flowrate, lbm/sec		Wall Temperature, °R Pump Inlet	
	LOX	H2	LOX	H2
No Insulation or Paint	1.0	1.0	181.4	40.3
LOX Insulation	1.0	1.0	179.8	40.3
LOX Insulation with Nickel Coating Effective as Reflector*	1.0	1.0	172.7	39.0
Thermal Control Paint, No Insulation**	1.0	1.0	169.5	38.6
No Insulation or Paint	4.04	4.04	169.5	38.6
Recirculation Fluid Temperatures: LOX at 163.7 °R and H2 at 36.7 °R				

* Best case of absorptance equals 0.40. This and the previous case bound the potential effects of insulation.

** Absorptance equals 0.18.

Figure 37. Pump Inlet Thermal Control Results

TA3–0292Fig

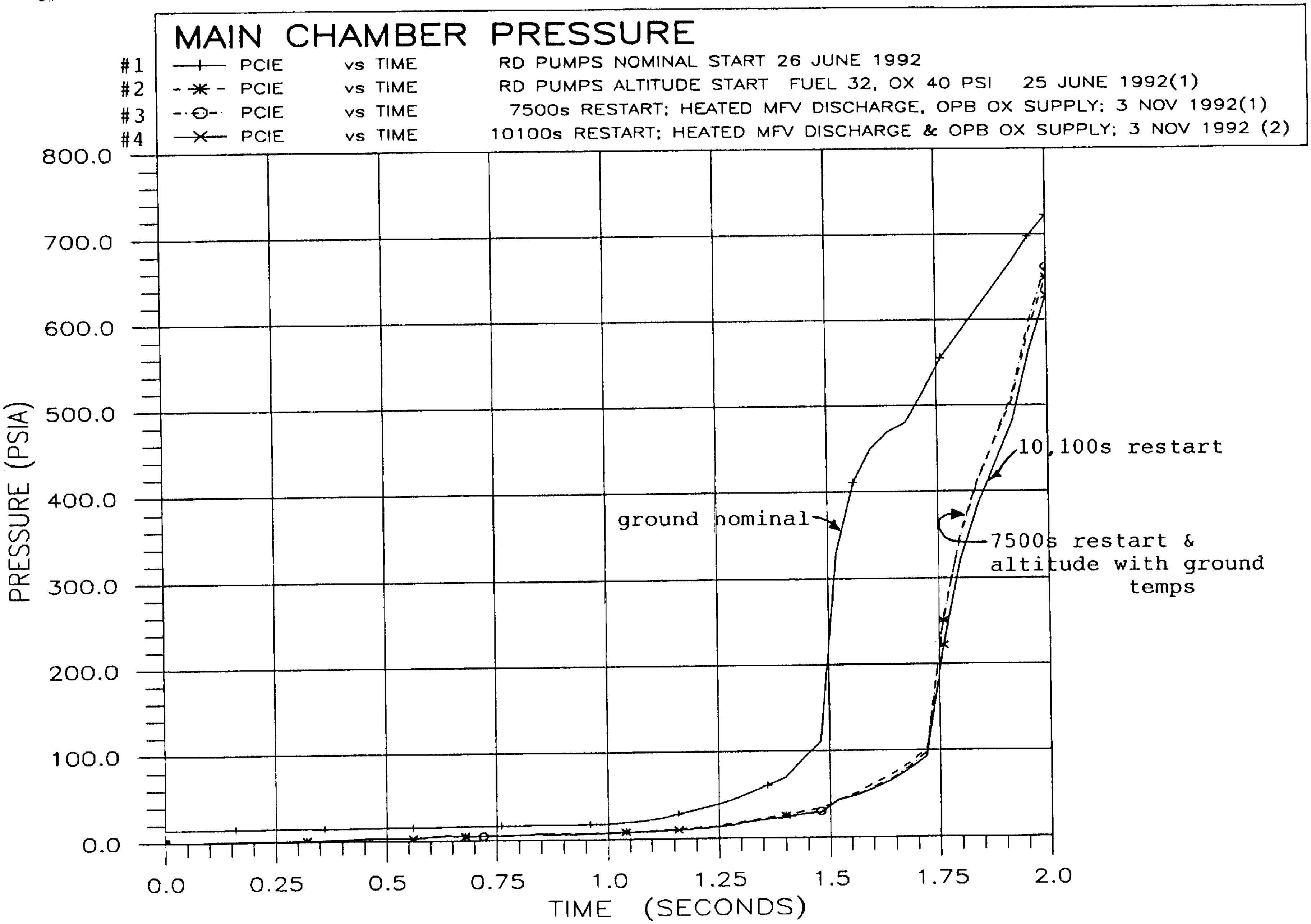


Figure 38. Summary of Start Results – Ground, Altitude, and Restart

TA3–0282Fig

- **Altitude Start Shown to be Feasible with Minimal Changes to Start Sequence**
  - **Valve Resequencing**
  - **ASI Orifice Changes**
  - **Inlet Pressures**
    - **LOX $\geq$ 40 psi**
    - **H2 $\geq$ 32 psi**

- **Orbital Restart Shown to be Feasible**
  - **Same Start Sequence as Altitude Start**
  - **Anytime After $\sim$ 2 Hours**
    - **1 lbm/sec Recirculation of LOX and H2 for $\sim$ 90 Minutes Prior to Restart**
    - **Thermal Control Paint on LOX Turbomachinery and Ducting**
    - **Component Heating Required**
      - **Main Fuel Valve Discharge/Coolant Control Valve**
      - **Oxidizer Preburner Oxidized Supply Duct**
  - **Restart as Soon as One Hour Possible with Additional Component Heating**

Figure 39. SSME Upper Stage Use Conclusions

TA3–0293Fig

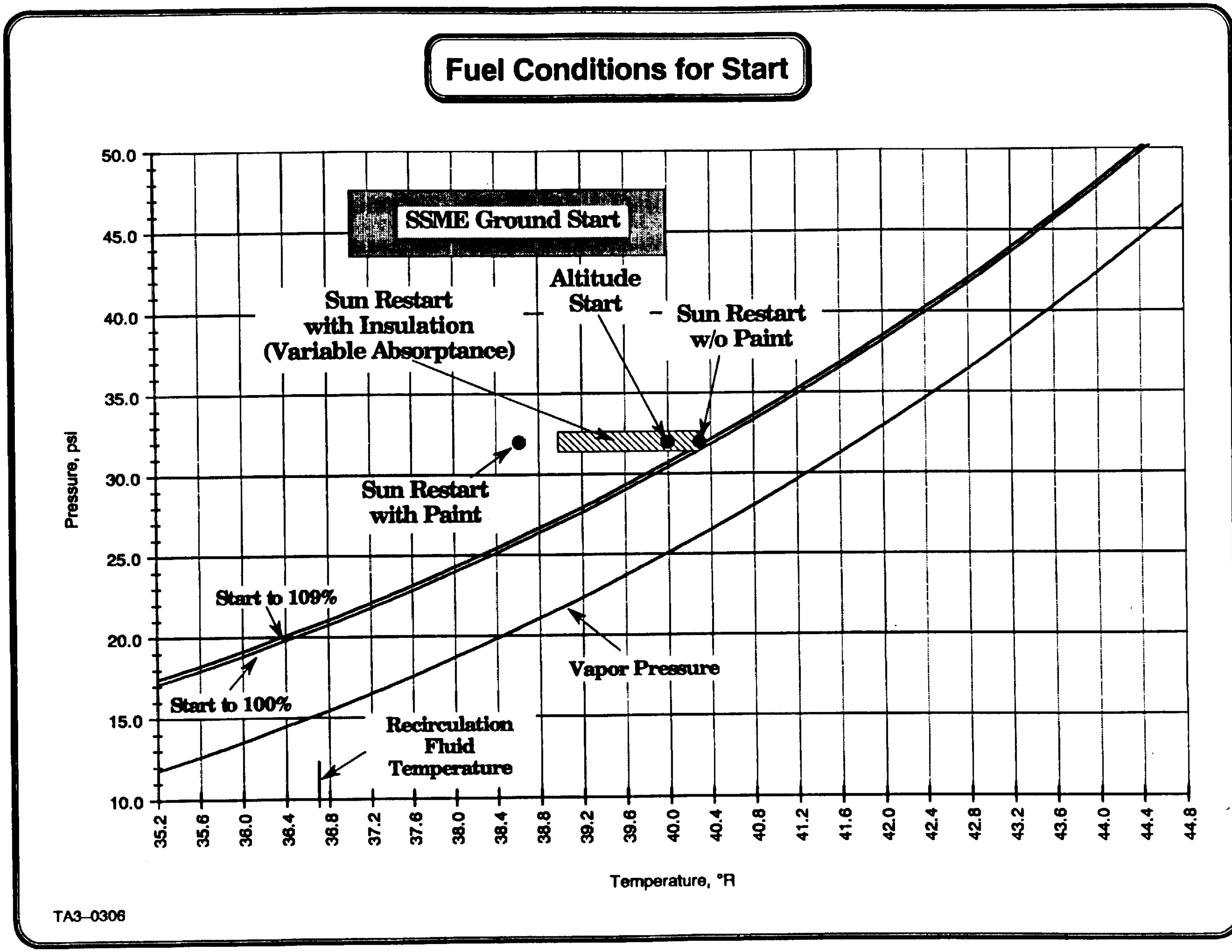


Figure 40. Fuel Conditions for Start

TA3–0306A2Fig

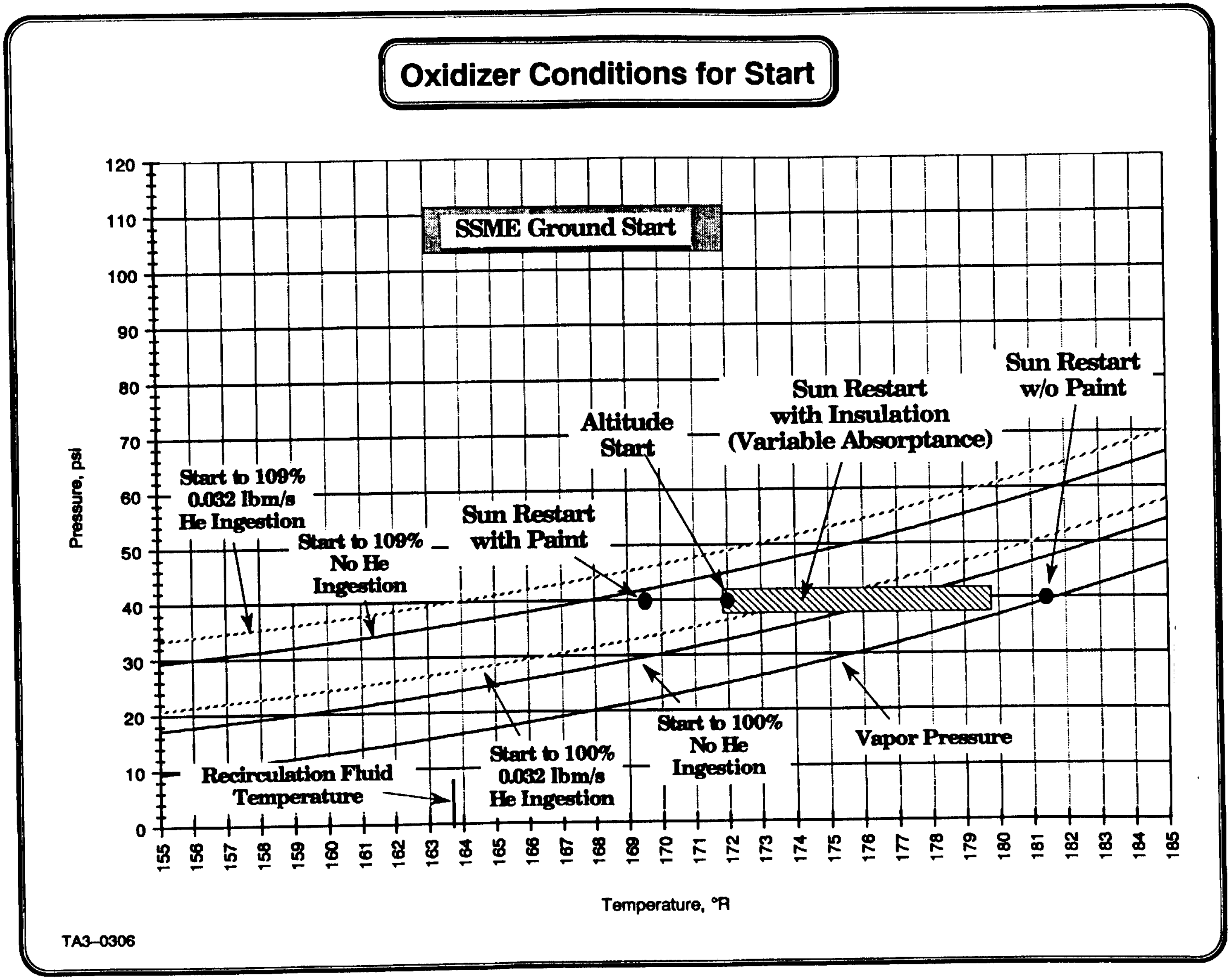


Figure 41. Oxidizer Conditions for Start

TA3–0306A4Fig

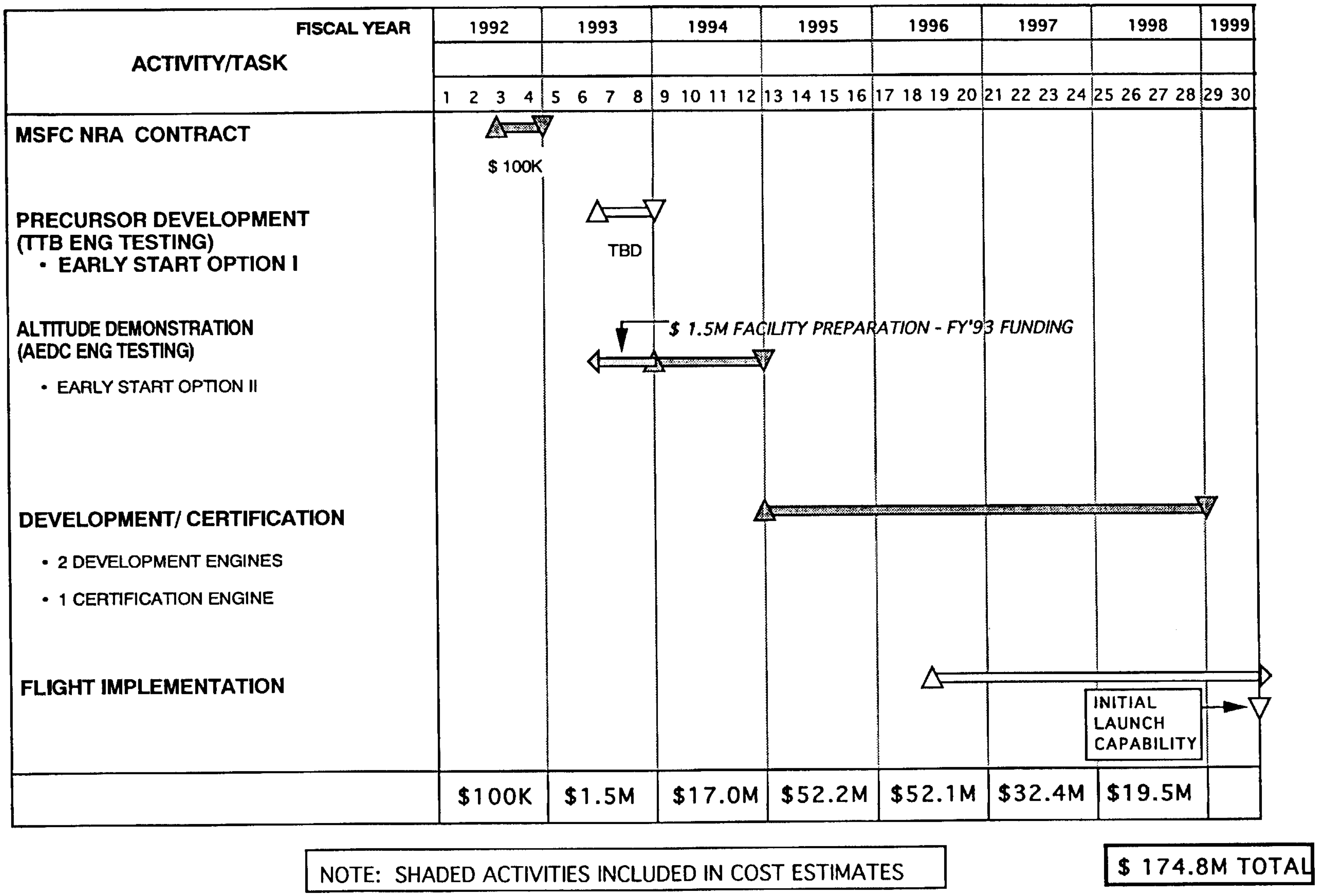


Figure 42. SSME Upper Stage Use Development Plan Cost and Schedule

TA3-0296Fig

CERs for Liquid Engines

Life Cycle Cost Elements

TOTAL ENGINE DEVELOPMENT COST TO TAXPAYER

PLUS

- Hardware
- Materials
- Manufacturing
- Manuf. Support
- Management
- Acceptance Test
- Fee
- Propellants
- Government Support

PRODUCTION

PLUS

- Engineering
- Hardware
- Testing
- Certif. Extension
- Progr. Mgmt.
- Fee
- Propellants
- Government Support, GFP

PREPLANNED PROGRAM IMPROVEMENT (P^3I)

PLUS

- Non-Recurring Cost
- Recurring Cost
- Government Facilities
- Government Support, GFP

OPERATIONS & SUPPORT

PLUS

- Facility Decommissioning
- Hardware Disposal
- Other

DISPOSAL

Figure 43. Life Cycle Cost Elements

TA3–0346Fig

$$C_U = 0.2455K_1 (F_{VAC})^{0.56} \times A_p \times A_{Reuse} \times A_{IMP} \times A_{Rate} \times A_Q \times A_{CIM} \times A_{ESC} \times A_{FEE}$$

C_U = Unit production cost, M$ (base FY1992); Valid Range 20 to 2000 Klbs Vac. Thrust

K_1 = Cycle and Propellant dependent factor

K_1 = 0.93 for gas generator cycle, LOX/RP, expendable

2.15 for gas generator cycle, LOX/H_2, expendable

4.84 for staged combustion, LOX/H_2, reusable

F_{VAC} = Vacuum thrust, Klbs

A_p = Chamber pressure effect; Valid range $500 \leq Pc \leq 3000$ psi for gas generator cycle

$$A_p = 1.255 - 3.4 \times 10^{-4} P_c + 8.5 \times 10^{-8} P_c^2$$

Continued

Figure 44. Production Cost Model Algorithms

Factor	Value	Conditions for Factor Value
A_{REUSE}	1.0	Reusable LOX/LH_2 staged combustion engines with >50 mission capability
	1.0	Expendable LOX/RP and OX/LH_2 gas generator engines with <20 mission capability
	0.8	Expendable LOX/LH_2 staged combustion engines
	1.1	Reusalbe LOX/LH_2 or LOX/RP gas generator engines with >20 mission capability
A_{IMP}	1.0	No producibility improvement, i.e., historical manufacturing environment
	0.77	Design simplifications only, no manufacturing improvement
	0.60	Design simplification and manufacturing improvement for derivative, expendible engines
	0.3	Potential cost factor for clean sheet design of new, low performance, expendible engines with high producibility
A_{RATE}	$1.61(R)^{-0.1392}$	Production rate improvement (91.1% Crawford unit cost improvement curve)
		R = Production rate in units per year for $18 < R < 50$
	$2.74(R)^{-0.3219}$	Production rate improvement (80% Crawford unit cost improvement curve)
		for $3 \leq R < 18$ (R = 3 is minimum production rate)
A_Q	$(Q)^{-0.0589}$	Production quantity improvement (96% Crawford unit cost improvement curve)
		Q = Production quantity, units
A_{CIM}	1.0	Conventional manufacturing environment
	0.71	Fully Computer Integrated Manufacturing (CIM) for R > 50/yr; 0% government investment
	0.64	Fully Computer Integrated Manufacturing (CIM) for R > 50/yr; 50% government investment
	0.54	Fully Computer Integrated Manufacturing (CIM) for R > 50/yr; 100% government investment
A_{ESC}	-	See curve on next chart; escalation factor from NASA Code BA for new start program, dated 4/6/92
A_{FEE}	1.0	Unit production cost
	1.1	Unit production price with nominal 10% fee

Figure 44. Production Cost Model Algorithms (Cont'd)

TA3–0368Fig

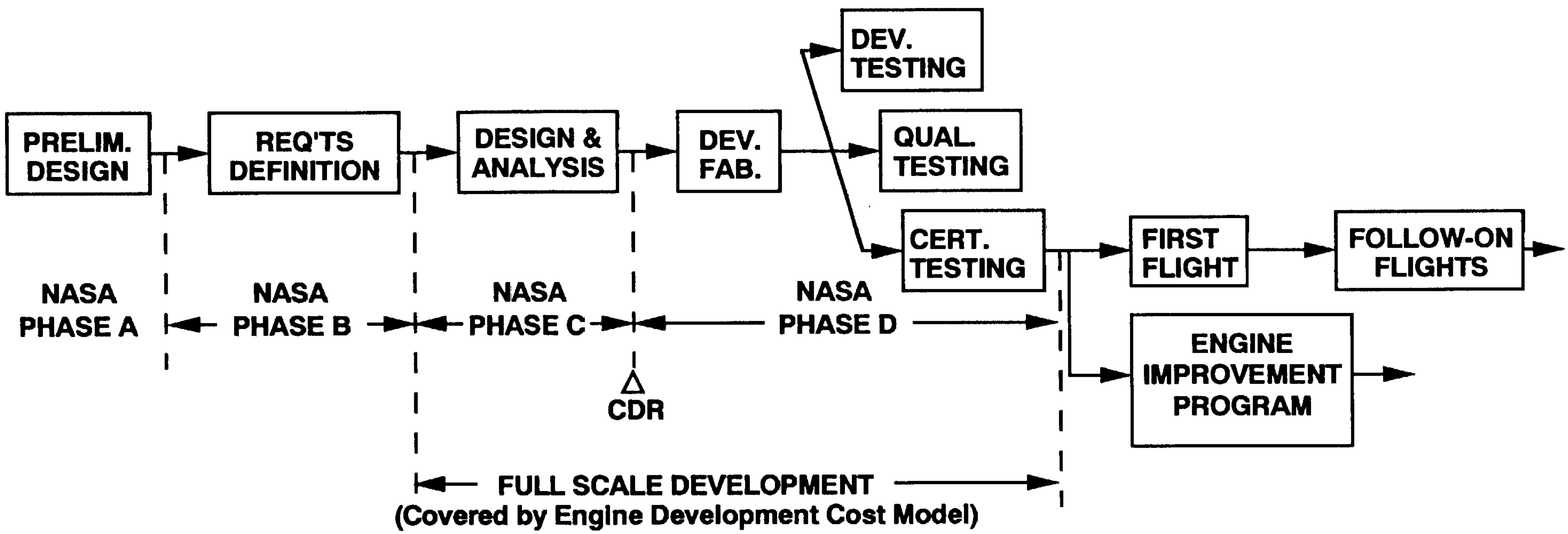
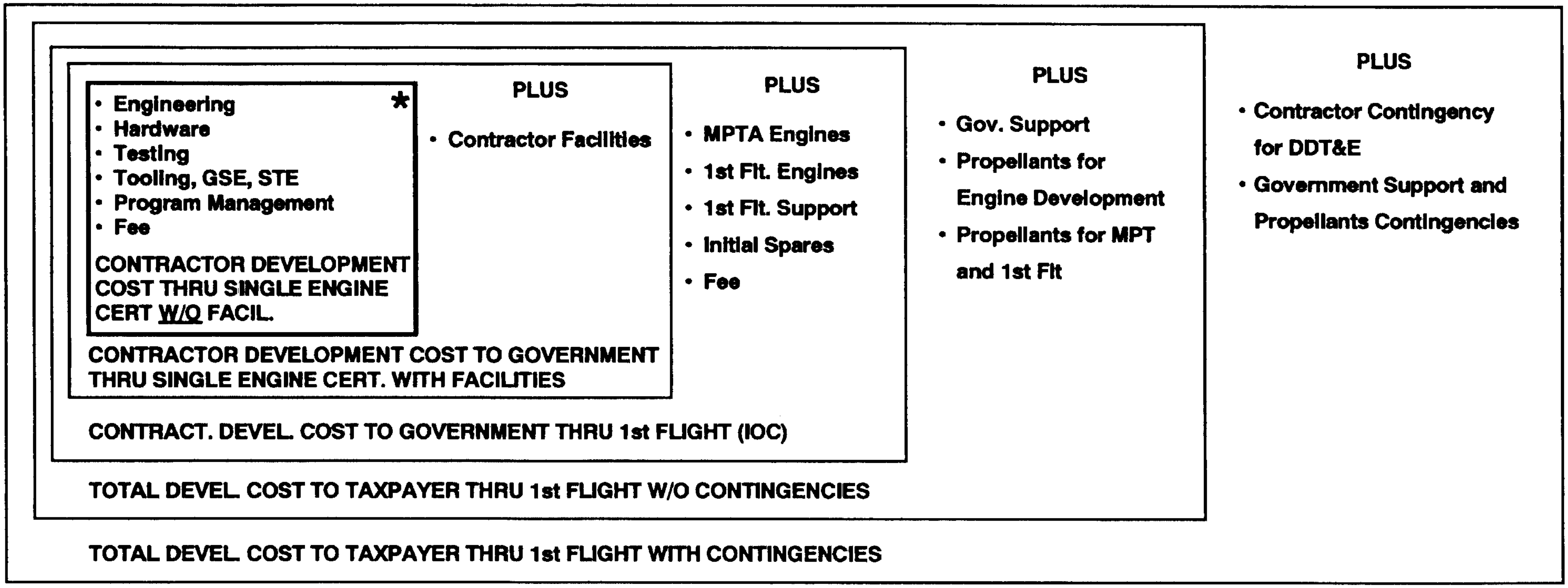


Figure 45. Engine Development Program Definition



* Included in Engine Development Cost Model

Figure 46. Engine Development Cost Composition

TA3-0392fig

INPUT PARAMETER	INPUT PARAM. RANGE	ALGORITHM	EQUATION #
CYPLX	0.6 - 1.2	(# OF TESTS TO CERT) = [571 (CYPLX)(ECMPLX)(PIF1)] + 10	1
ECMPLX	0.2 - 1.9		
PIF1	0.2 - 4.2		
		(# OF ENGINES) = [1 /TPE] + 1	2
		TPE = 27 IF (CYPLX)(ECMPLX)(PIF1) < 1.0	
		TPE = 38 IF (CYPLX)(ECMPLX)(PIF1) ≥ 1.0	
DET	3 - 5	(DEV. ENG. UNIT COST) = 1.14 (TFU @ R_1)	3
TFU		R1 = 2 /DET	
		(DEV. HDW. COST) = 2 * 3	4
TFRQ	10 - 30	(TEST LABOR COST) = 1 * [0.205 (30/TFRQ)(PIF2)]	5
PIF2	0.2 - 1.0		
PIF3	0.3 - 1.0	(DES. ENGR. LABOR COST) = 500 (CYPLX)(ECMPLX)(PIF3)	6
R2	1.0 - 1.33	(TOOL., GSE, STE COST) = 150 R2 (TAVAIL)(TIF)	7
TIF	0.6 - 1.0		
TAVAIL	0.25 - 1.0		
		(PRGRM. MGMT. COST) = 0.03 [4 + 5 + 6 + 7]	8
		(TOTAL DEV. PROGR. COST) = [4 + 5 + 6 + 7 + 8] * FEE FACTOR	9

Figure 47. Summary of Development Cost Model Algorithms

TA3-0408fig

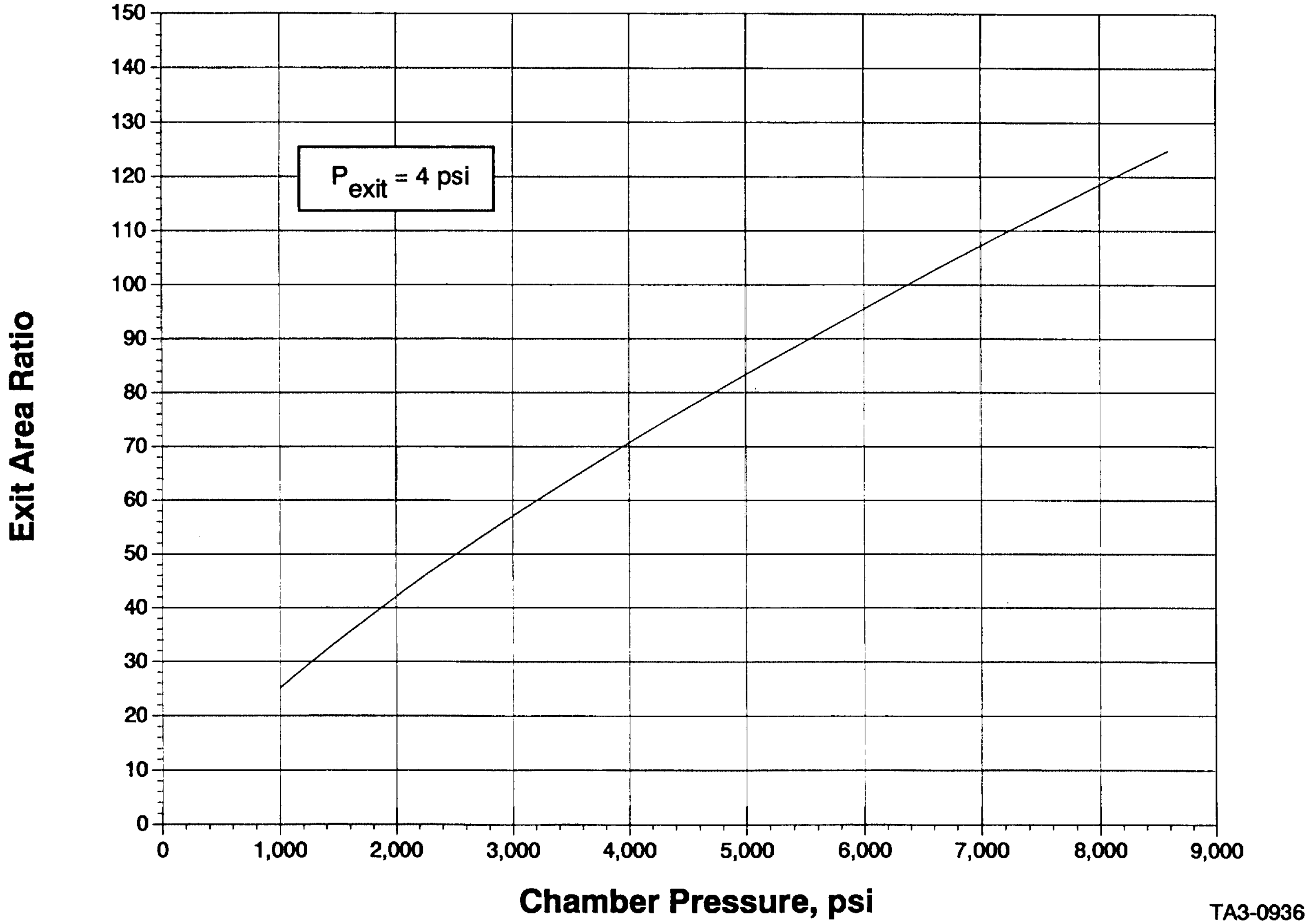


Figure 48. Area Ratio Versus Chamber Pressure

- All Engines Calculated for 421,000 lbf Sea Level Thrust
- Combustion Efficiency
  - 0.995 at MR = 6
  - 0.985 at MR = 7, 10
- Turbomachinery
  - Turbine Operating Temperatures
    - Fuel
      - 2,500 °R — Si_3N_4
      - 1,000 – 1,900 °R — Astroloy
      - 1,000 °R — Al
    - Oxidizer
      - 2,500 °R — Si_3N_4
      - 1,100 – 1,900 °R — Inco
  - Pumps
    - Use Boost Pumps
    - Use Kick Pumps Where Applicable
    - Fuel
      - 1 – 6 Stages (Al, Ti)
    - Oxidizer
      - 1 – 4 Stages (Inco)
- Fixed Bell Nozzle
  - Exit Area Ratio Sized for P_{exit} = 4 psi
- All Regenerative Cooling with H_2
- No Throttling Requirement

TA3–0670

Figure 49. Baseline Configuration Parameters

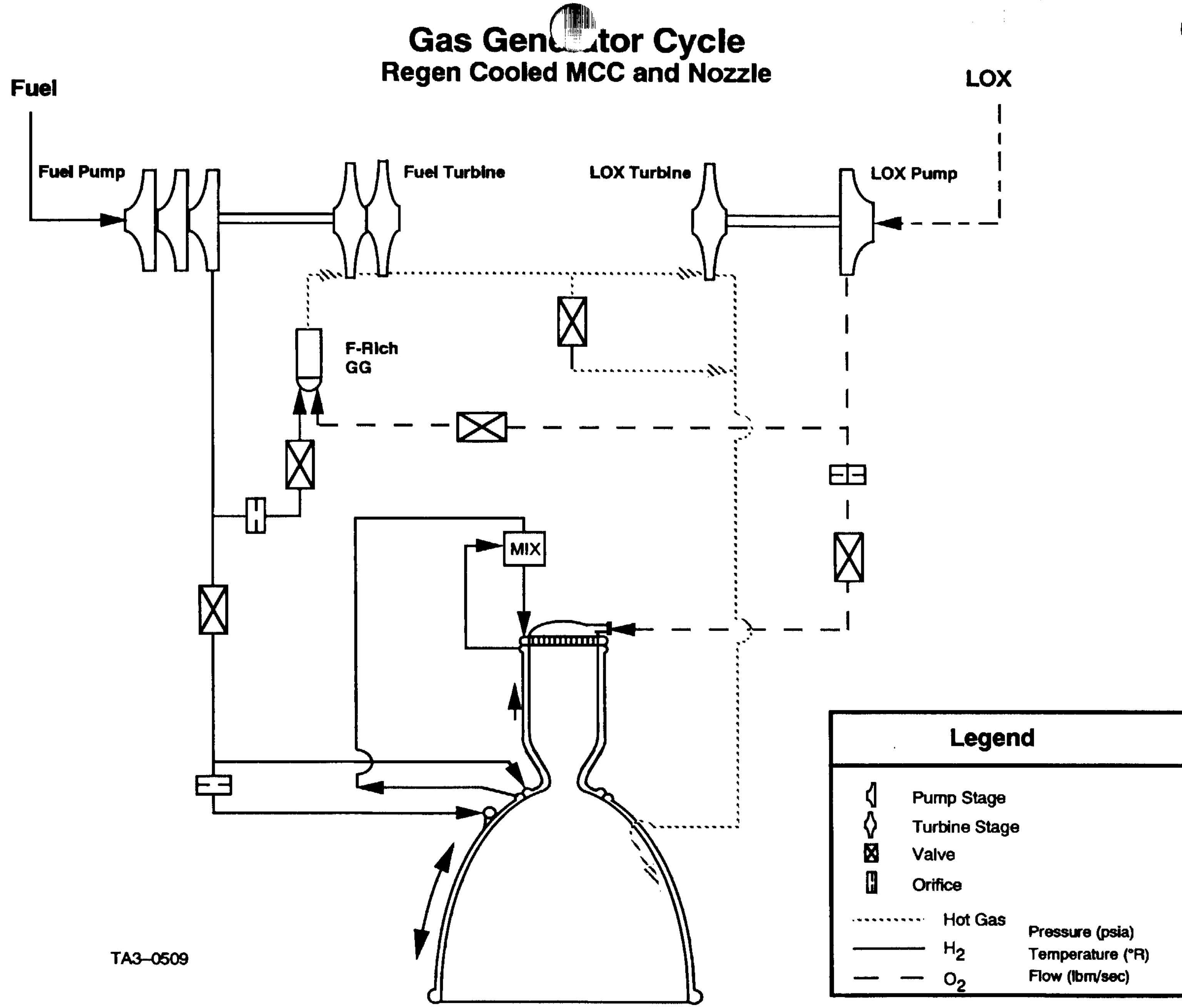


Figure 50. Gas Generator Cycle

FFSCC Mixed Preburner Engine

Regen Cooled MCC and Nozzle

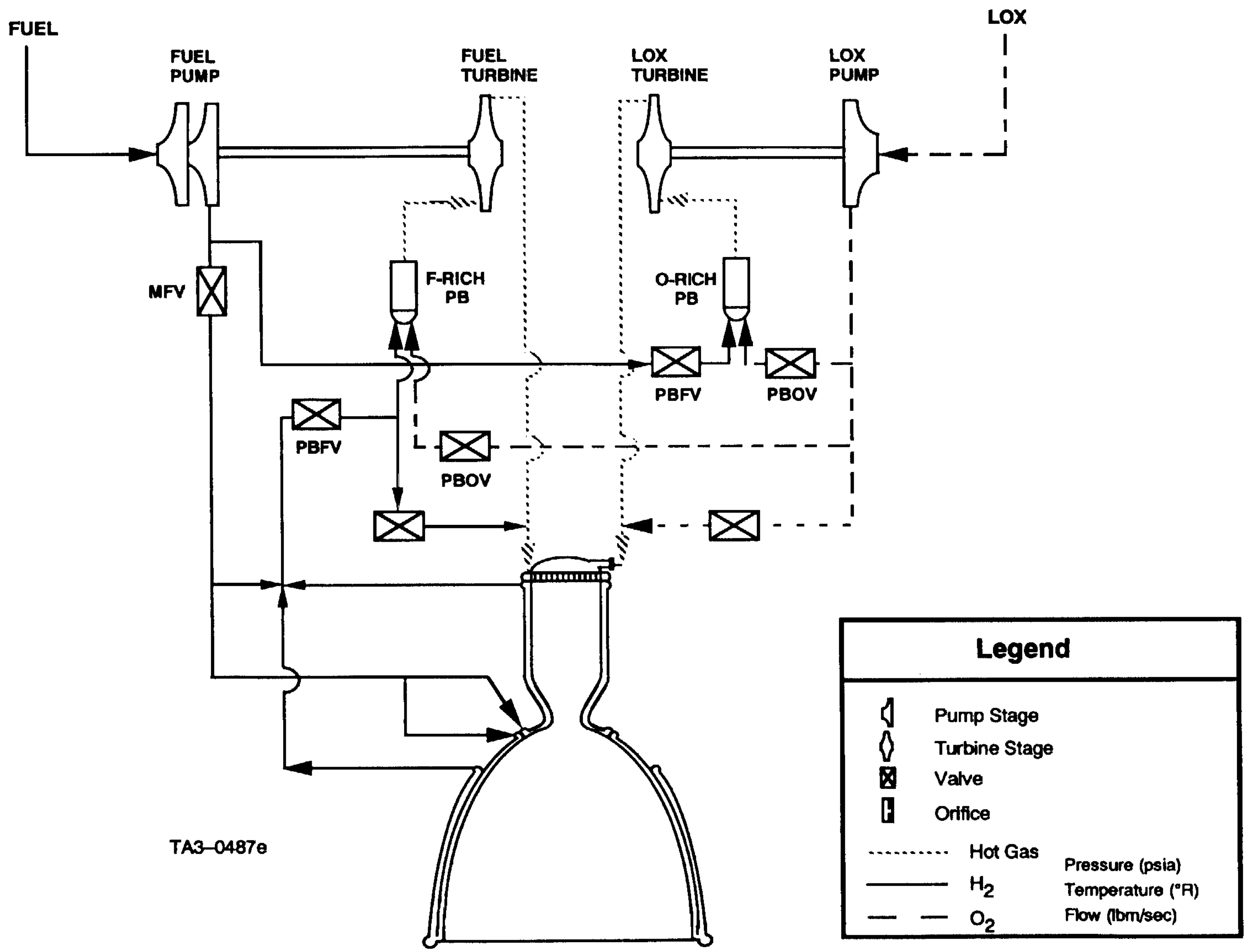


Figure 51. Mixed Preburners Full Staged Combustion Cycle (FFSCC)

SCC Dual Fuel-Rich Preburner Engine

Regen Cooled MCC and Nozzle

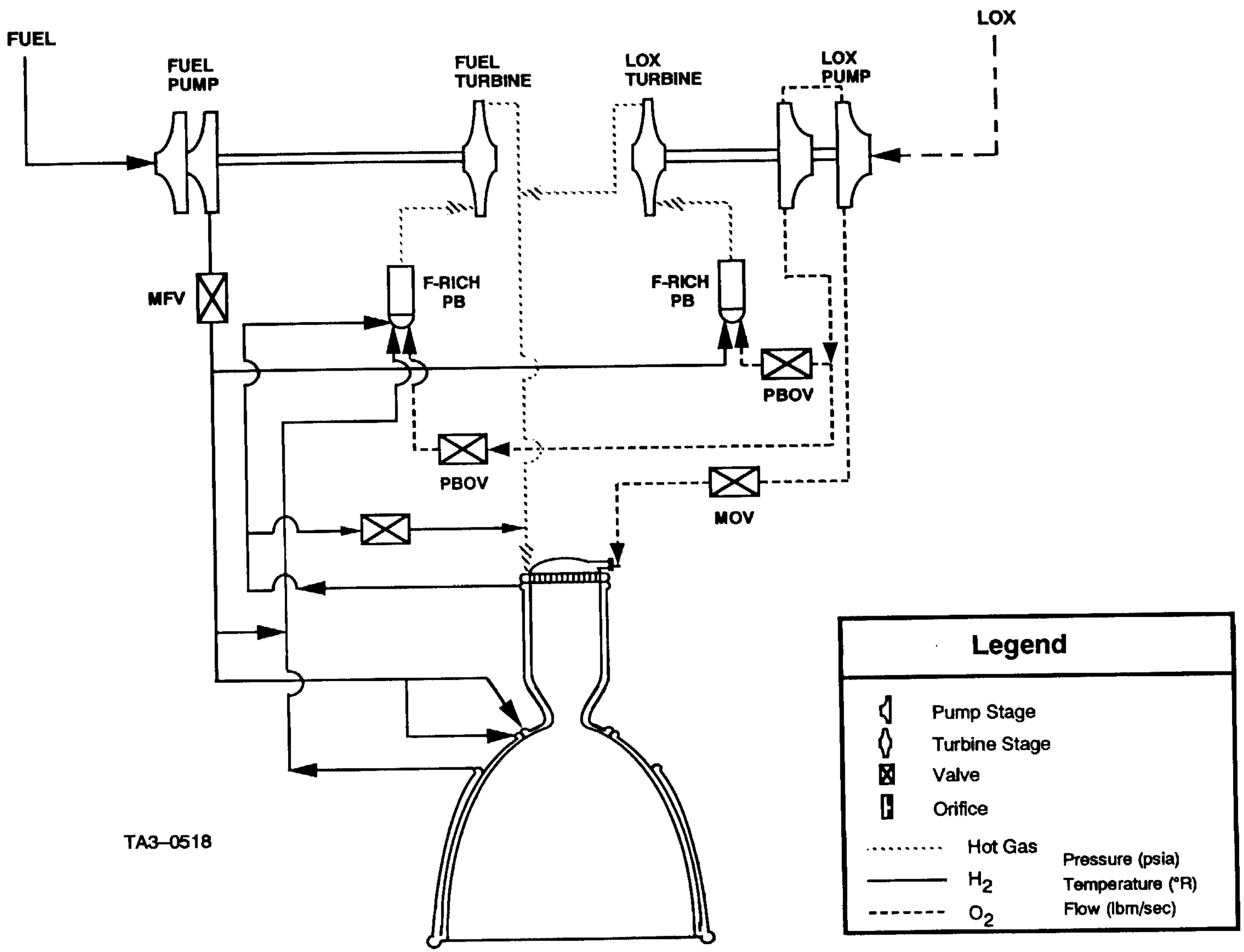


Figure 52. Staged Combustion Cycle (SCC)

Hybrid Cycle Engine
(Fuel Side Preburner, Ox Side Expander)
Regen Cooled MCC and Nozzle

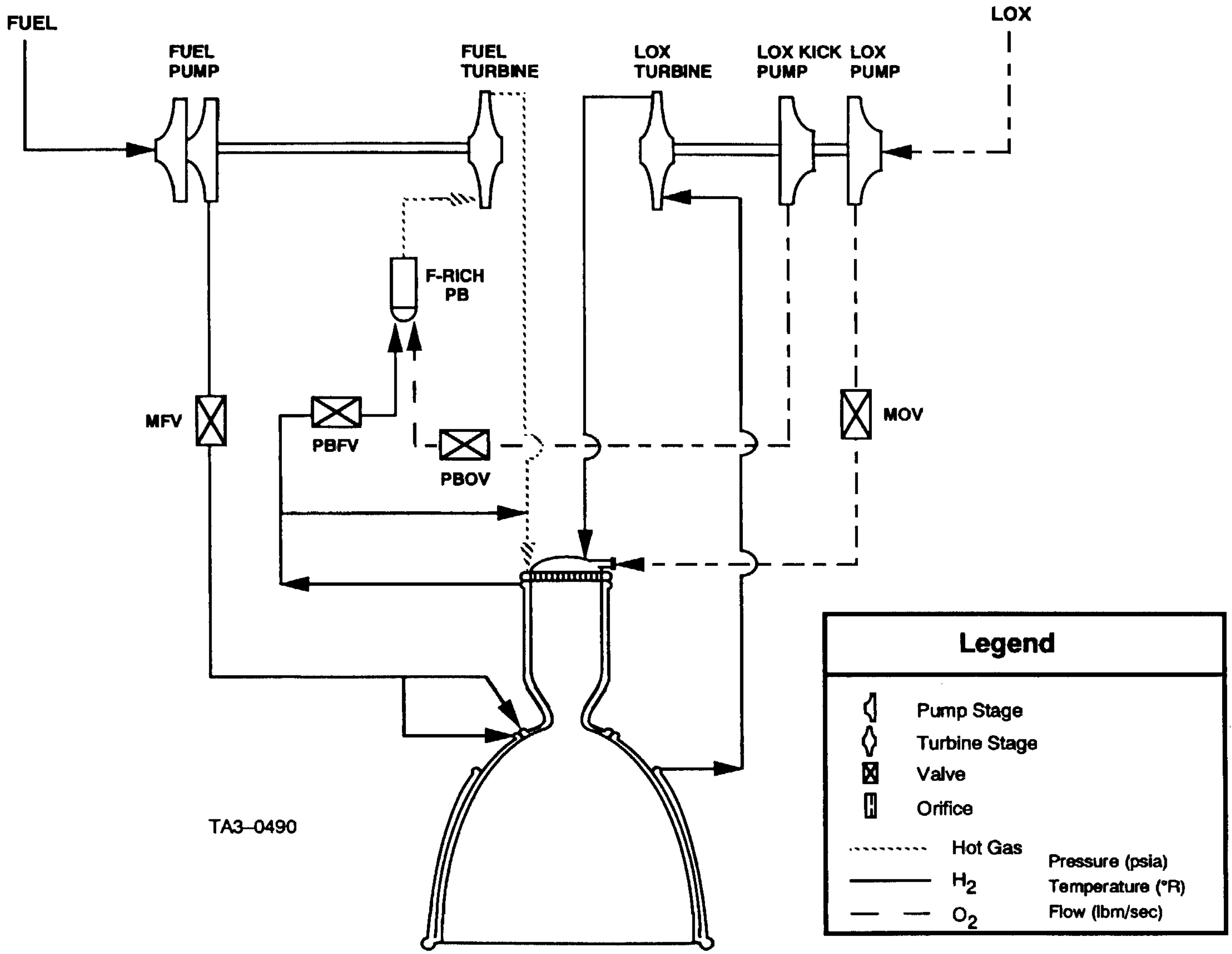


Figure 53. Hybrid Cycle

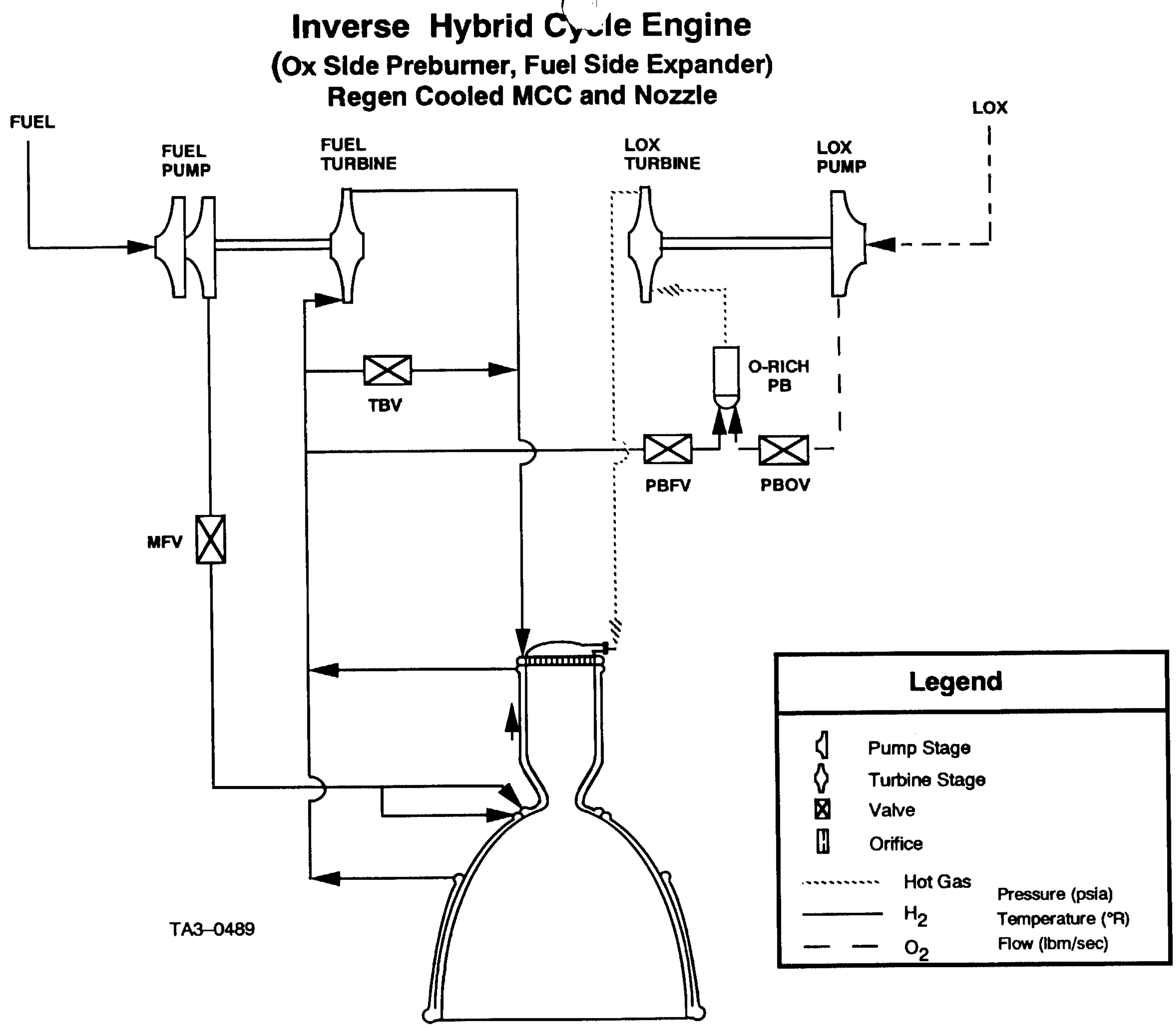


Figure 54. Inverse Hybrid Cycle

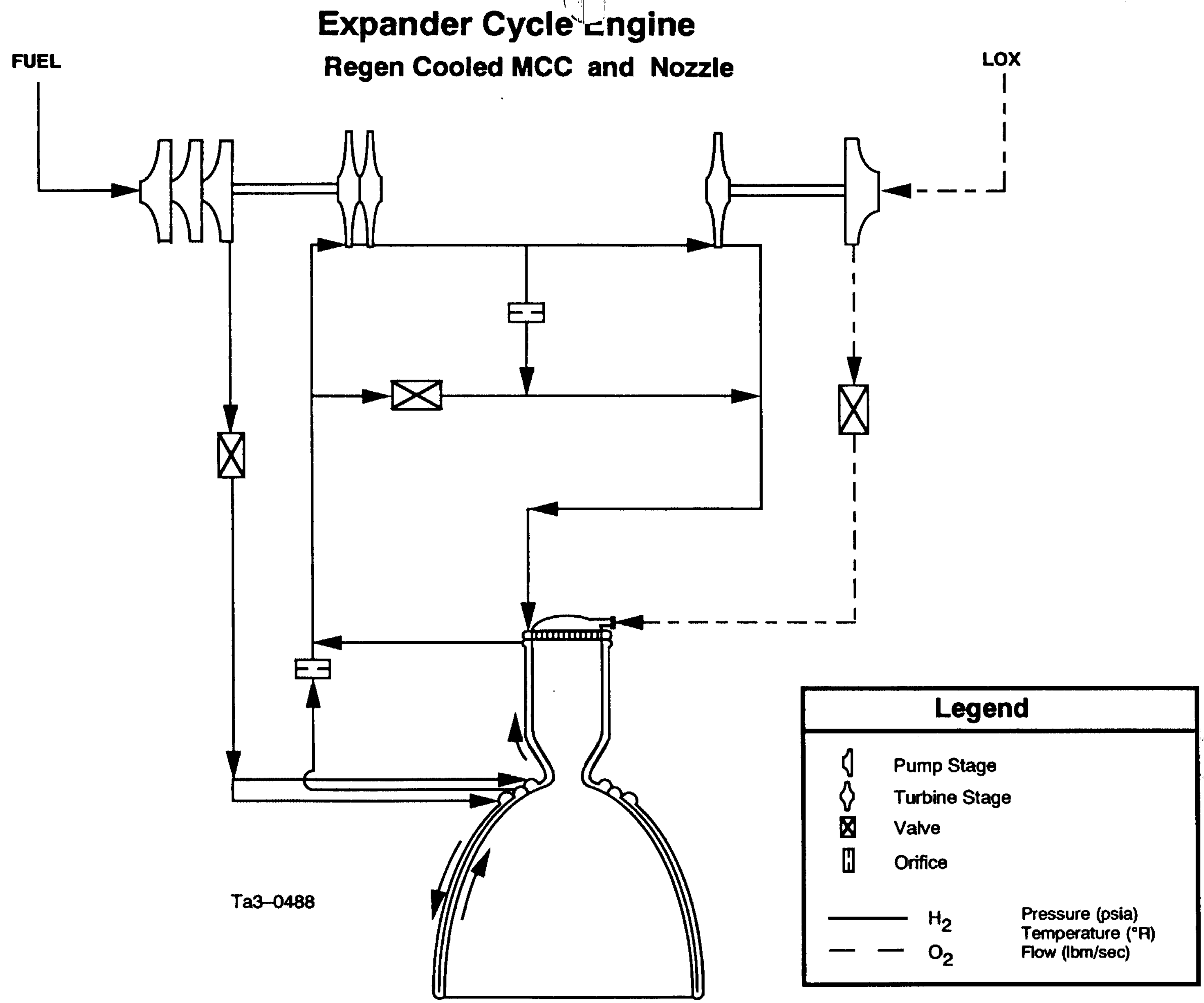


Figure 55. Expander Cycle

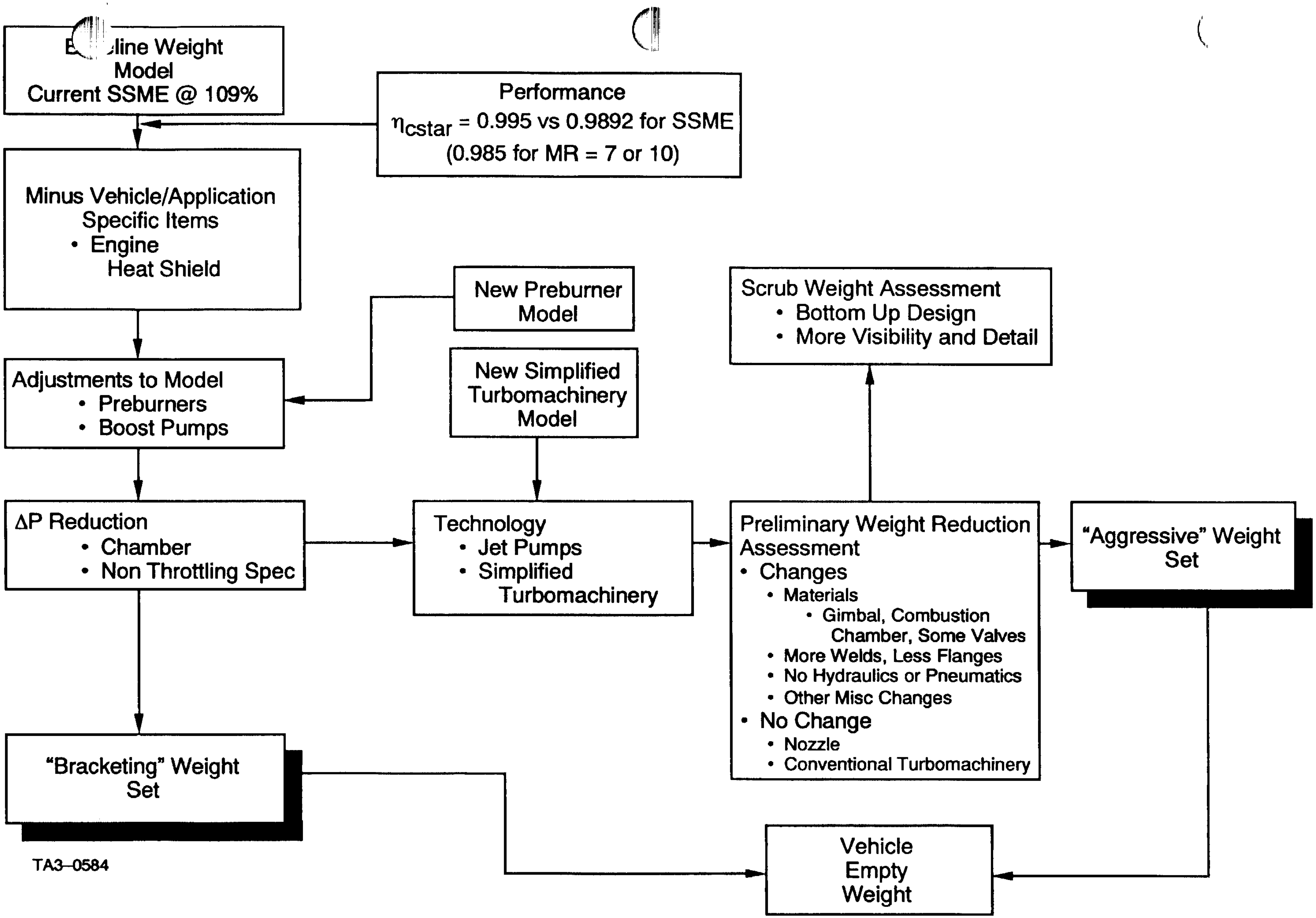


Figure 56. Engine Weight Generation Methodology

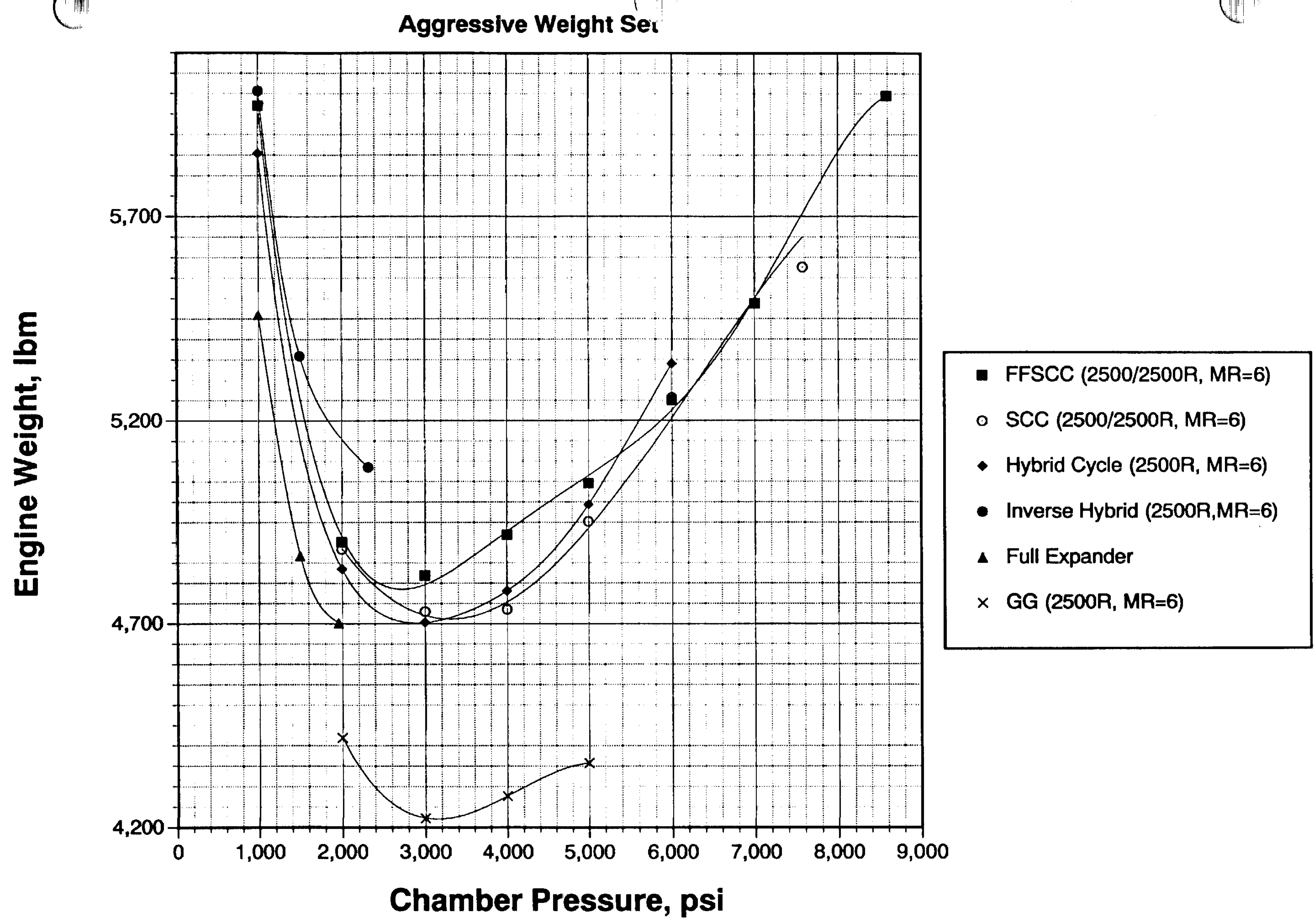


Figure 57. Engine Weights (Aggressive Set)

TA3–0918

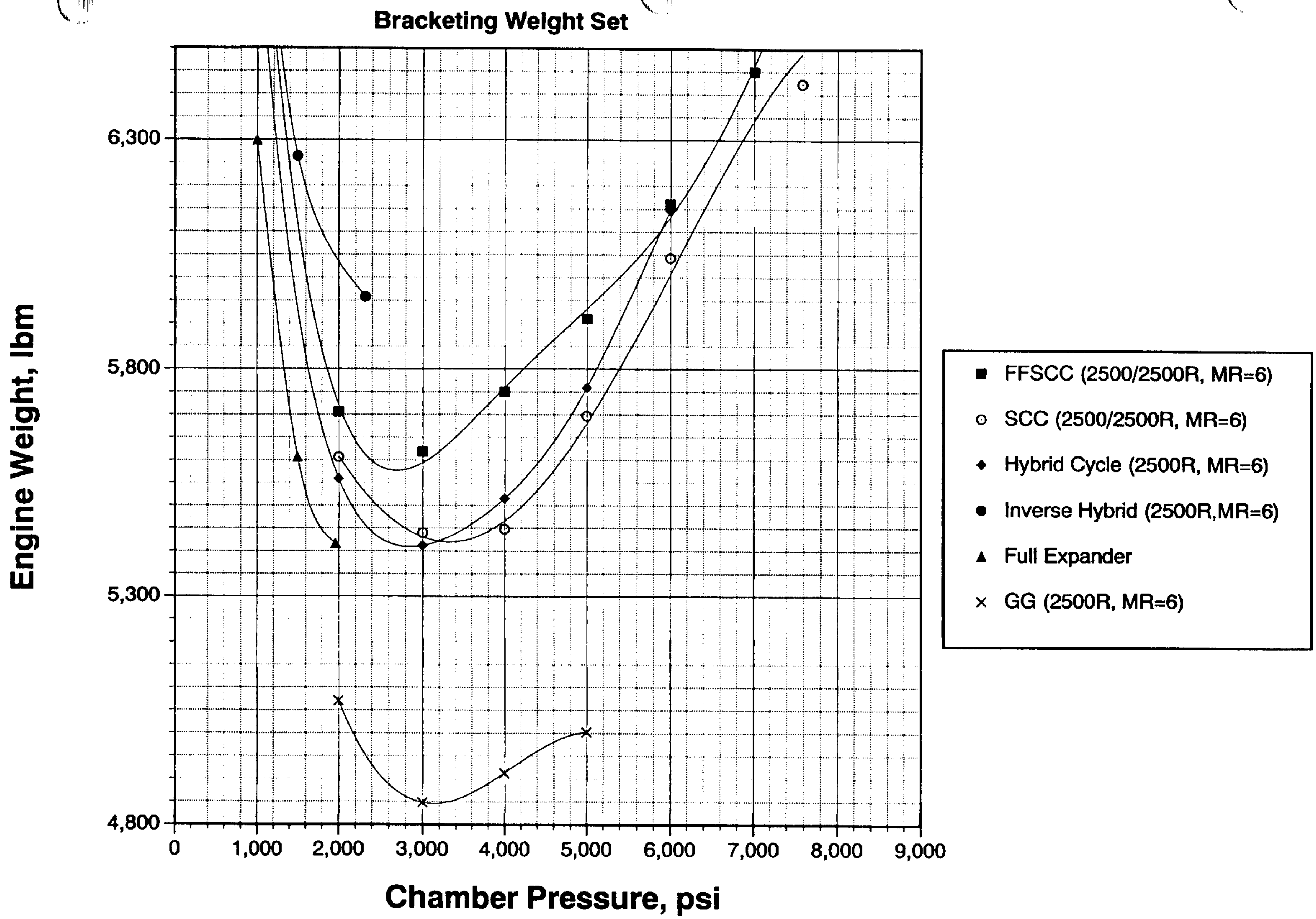


Figure 58. Engine Weights (Bracketing Set) TA3–0919

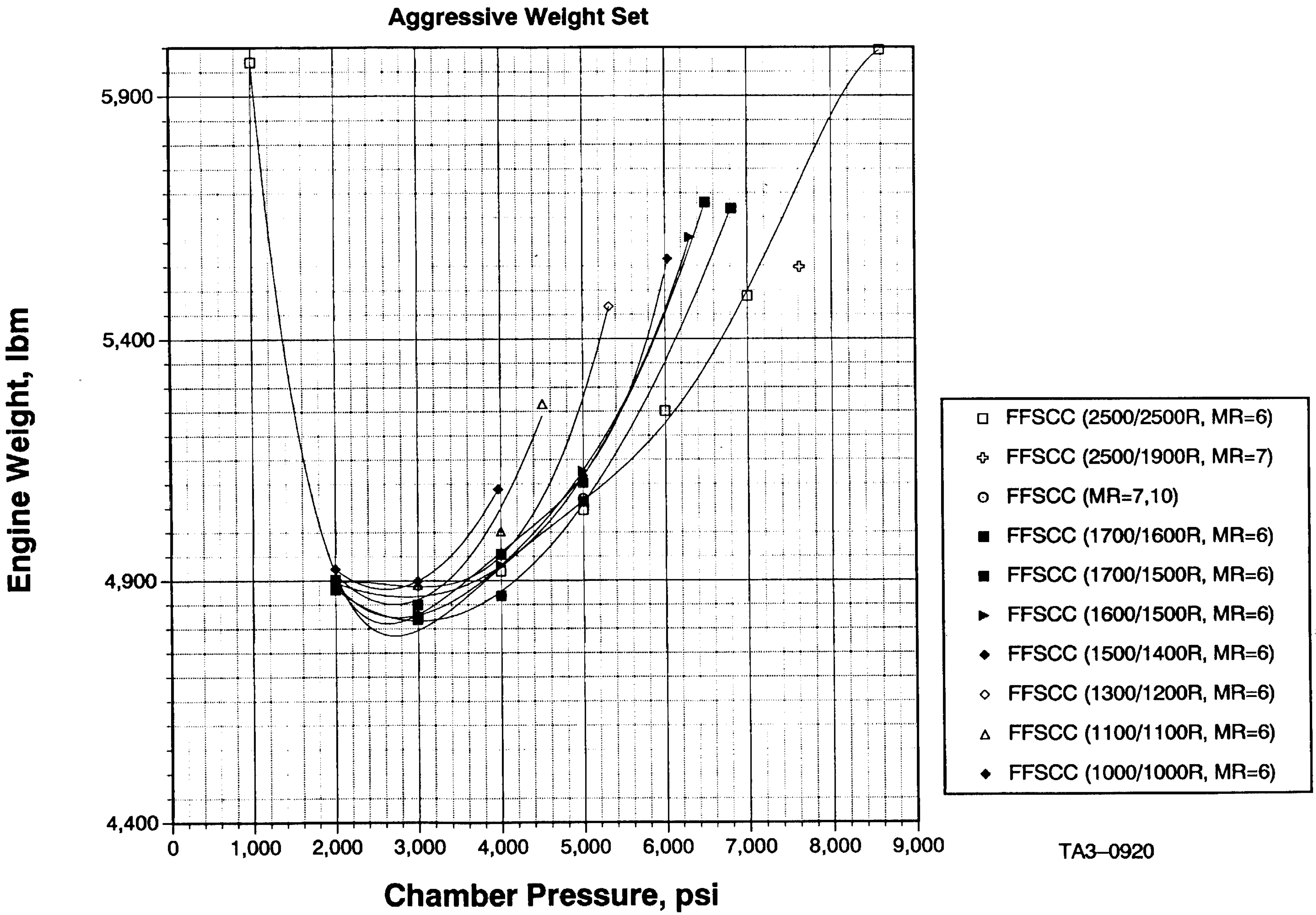


Figure 59. Engine Weights vs Turbine Temperature – FFSCC

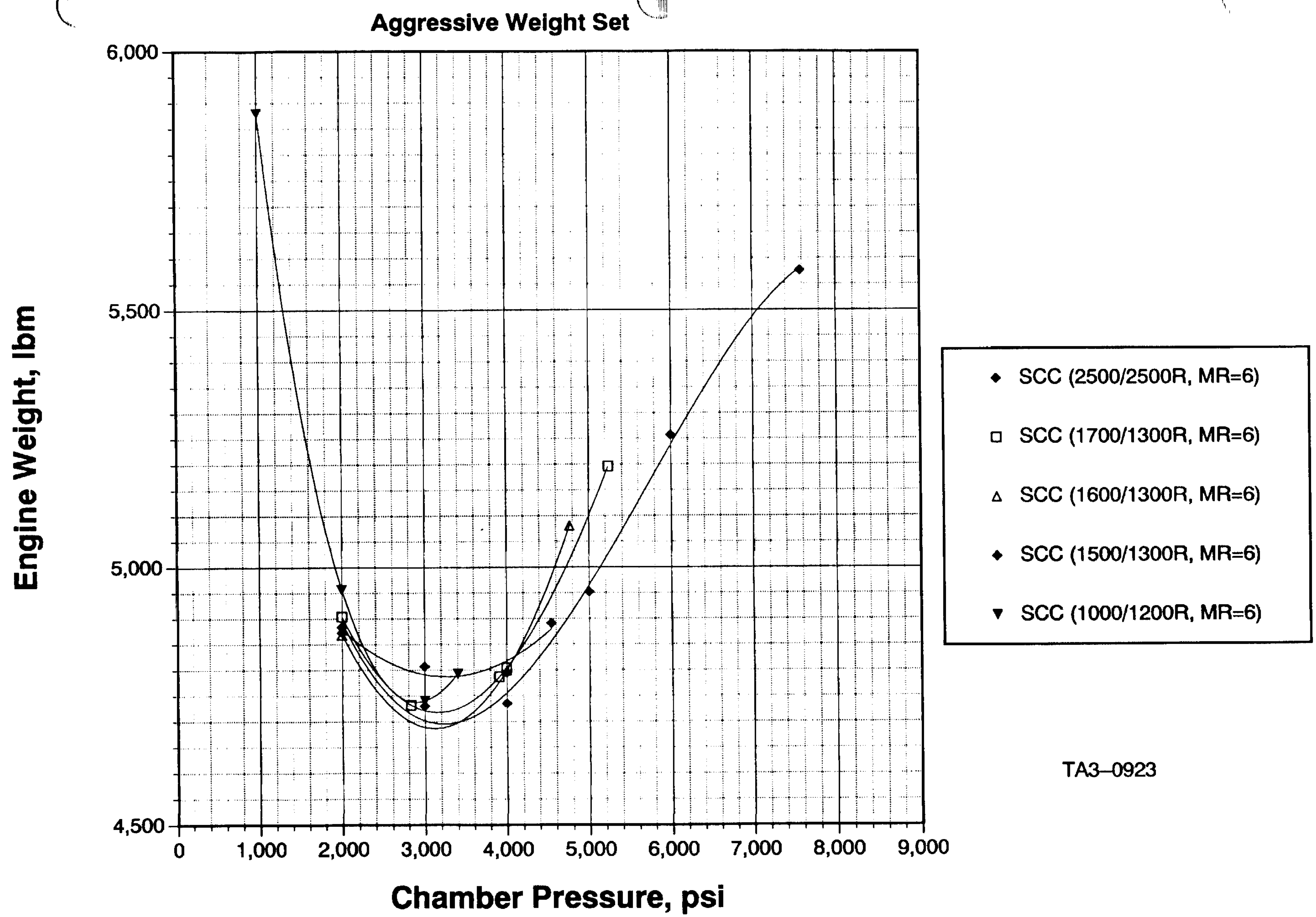


Figure 60. Engine Weights vs Turbine Temperature – SCC

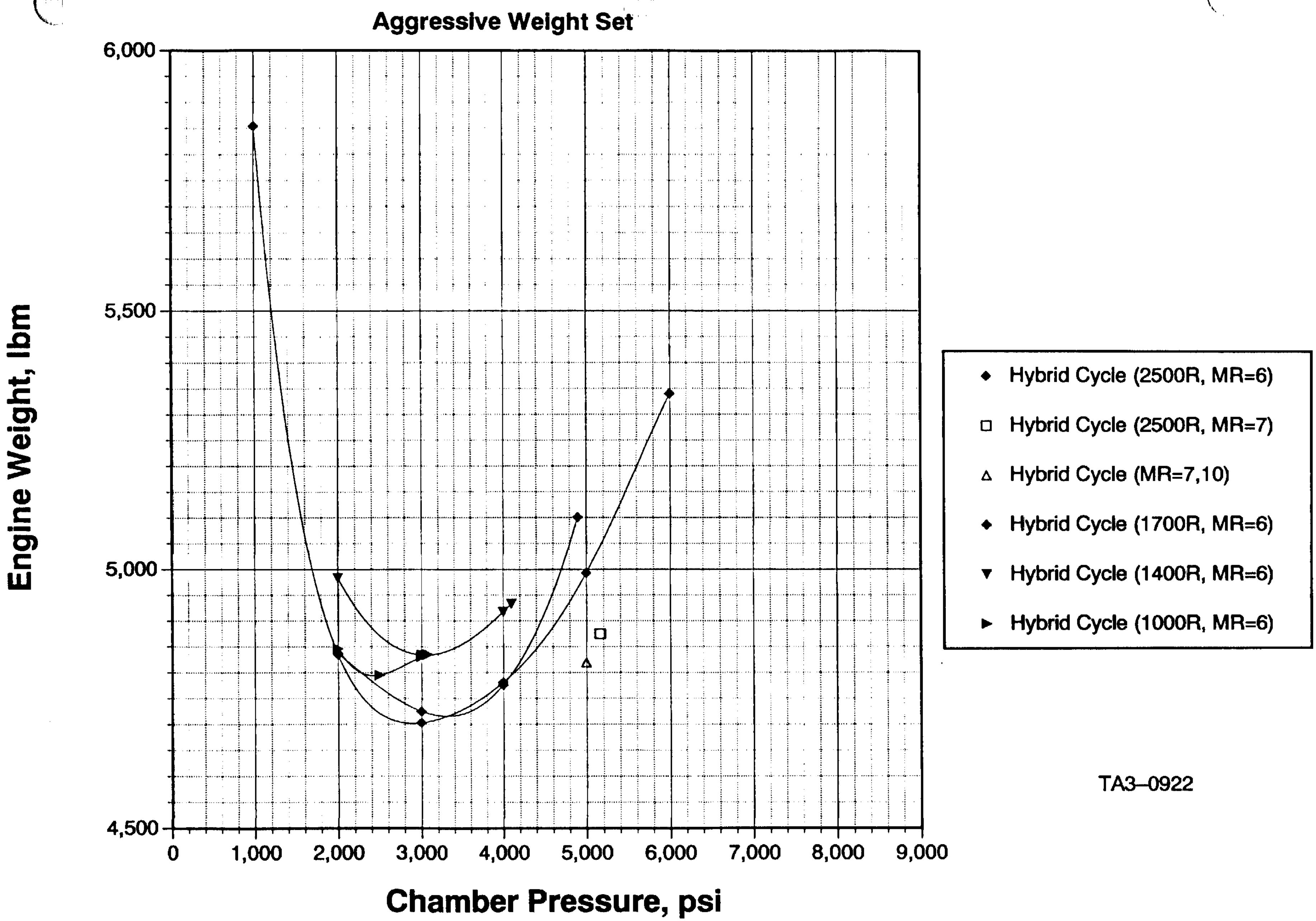


Figure 61. Engine Weights vs Turbine Temperature – Hybrid Cycle

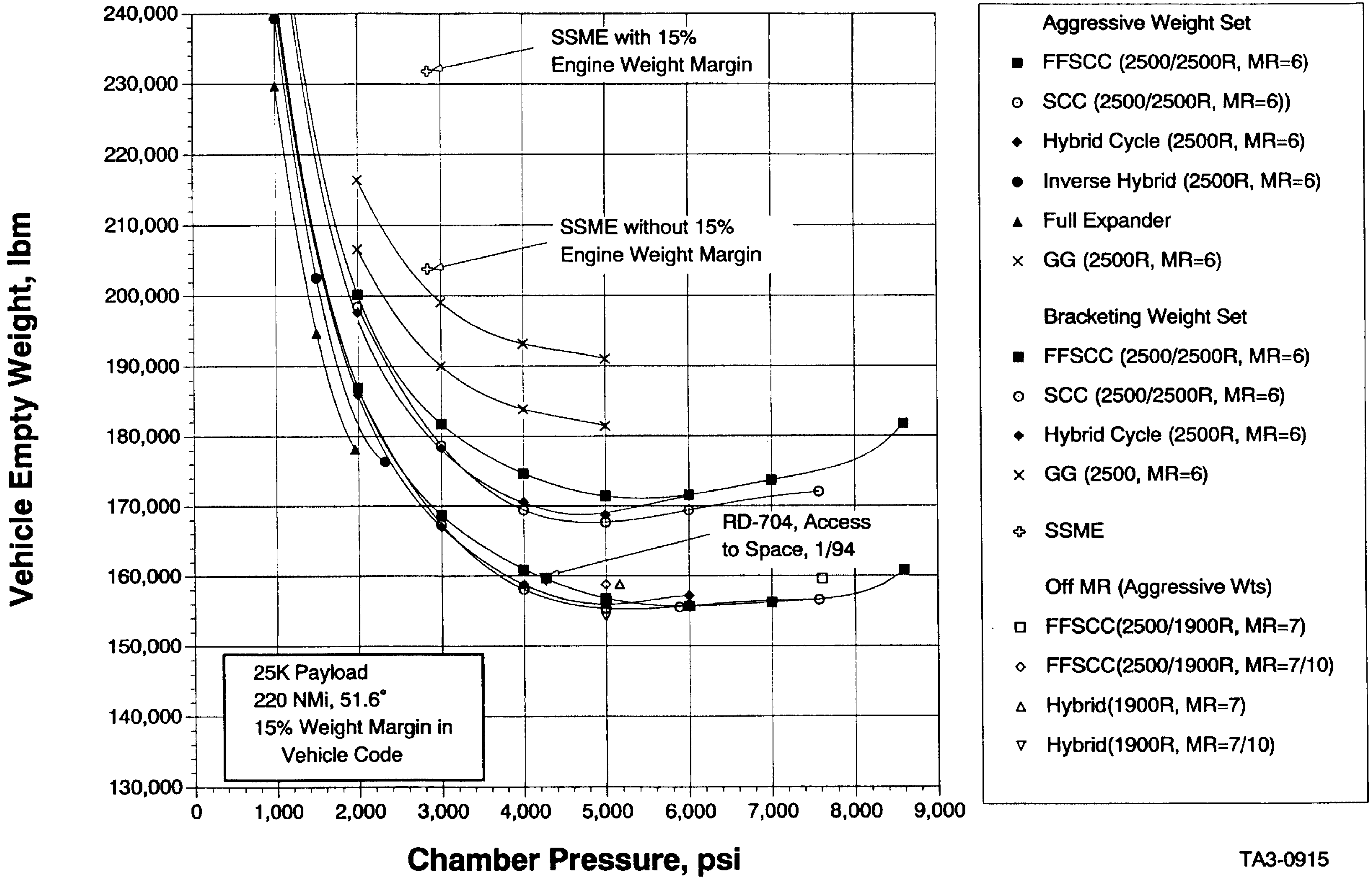


Figure 62. SSTO Performance of Advanced O_2/H_2 Engines

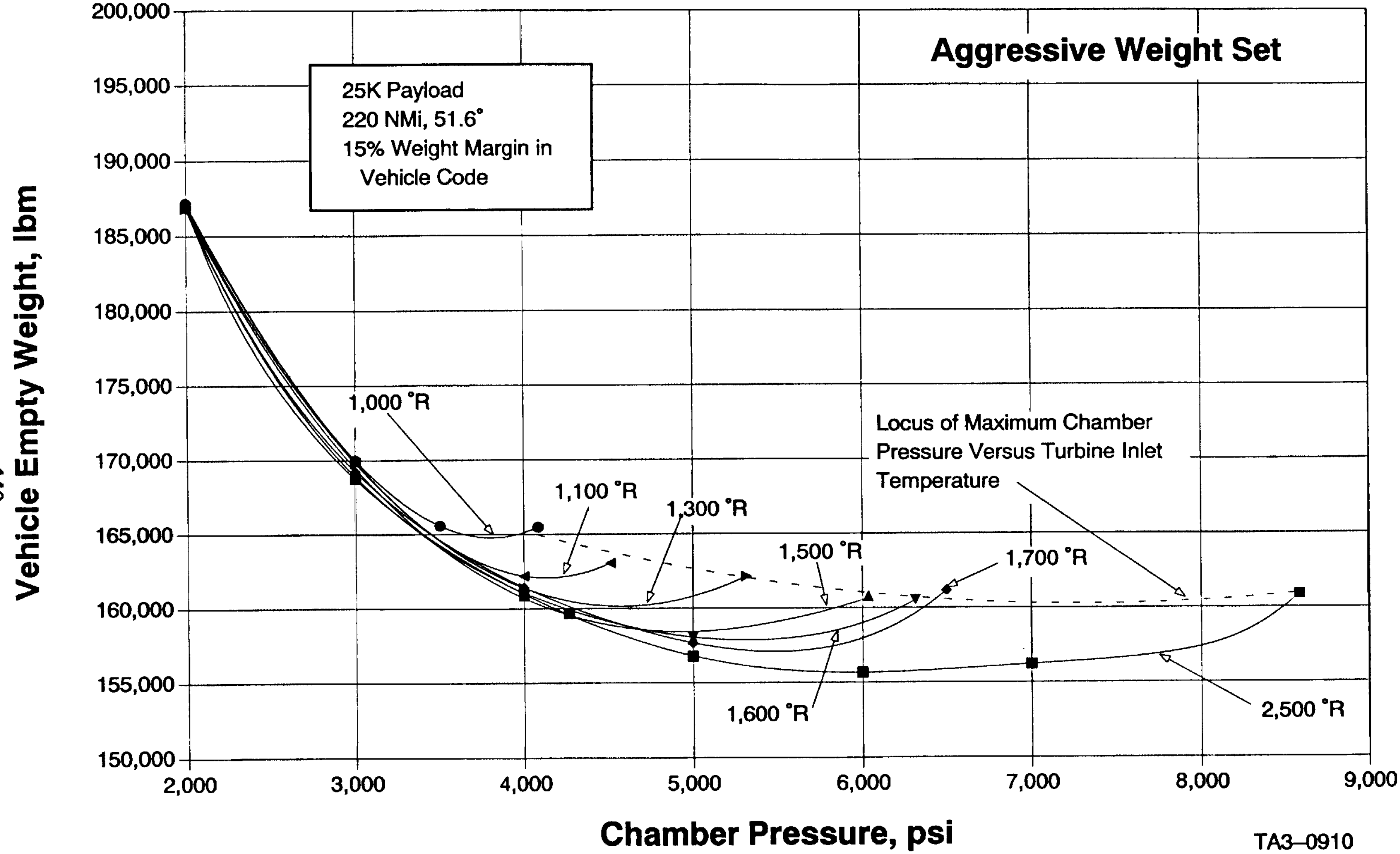


Figure 63. Effect of Fuel Turbine Inlet Temperature on SSTO Performance – FFSCC

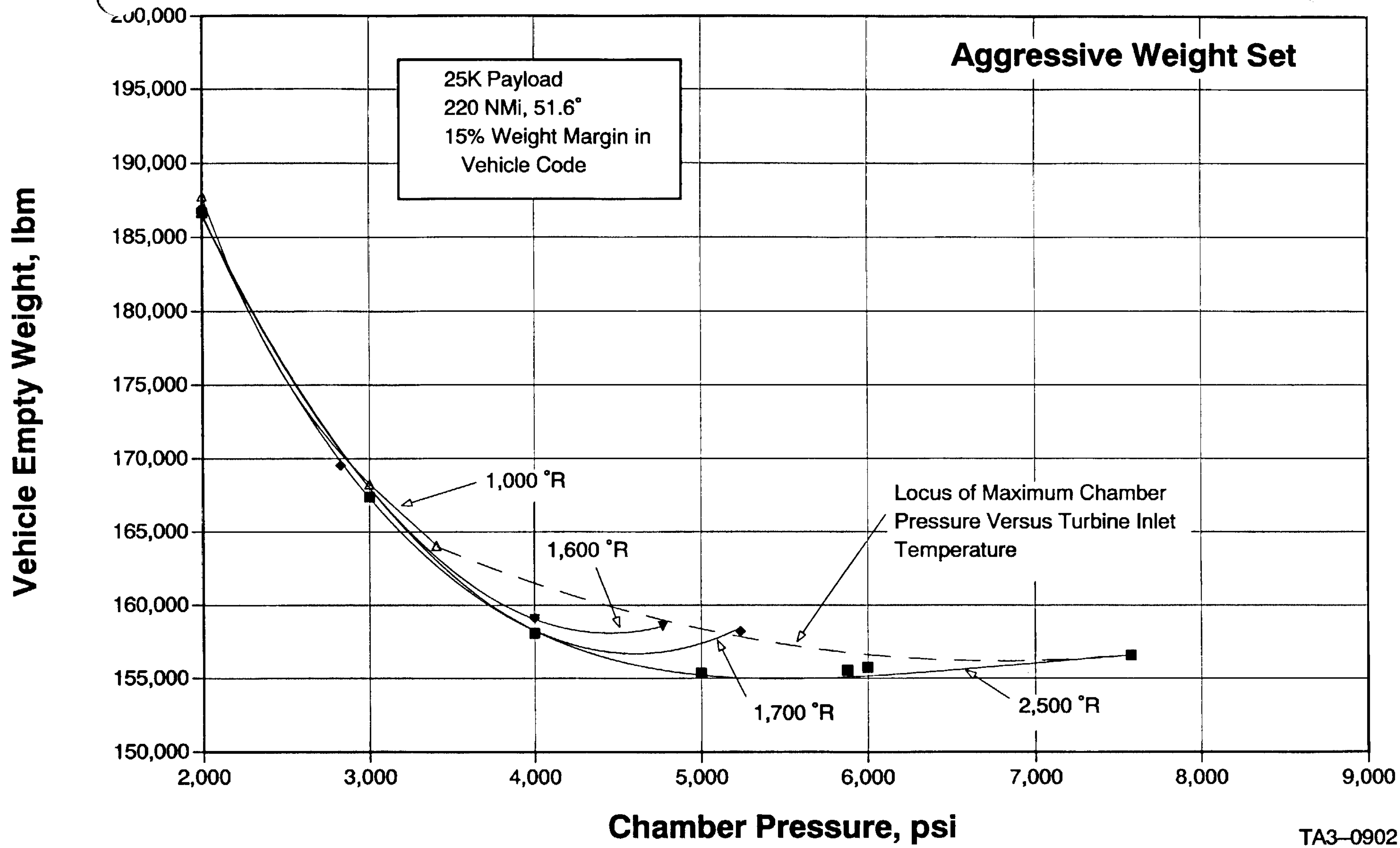


Figure 64. Effect of Fuel Turbine Inlet Temperature on SSTO Performance – SCC

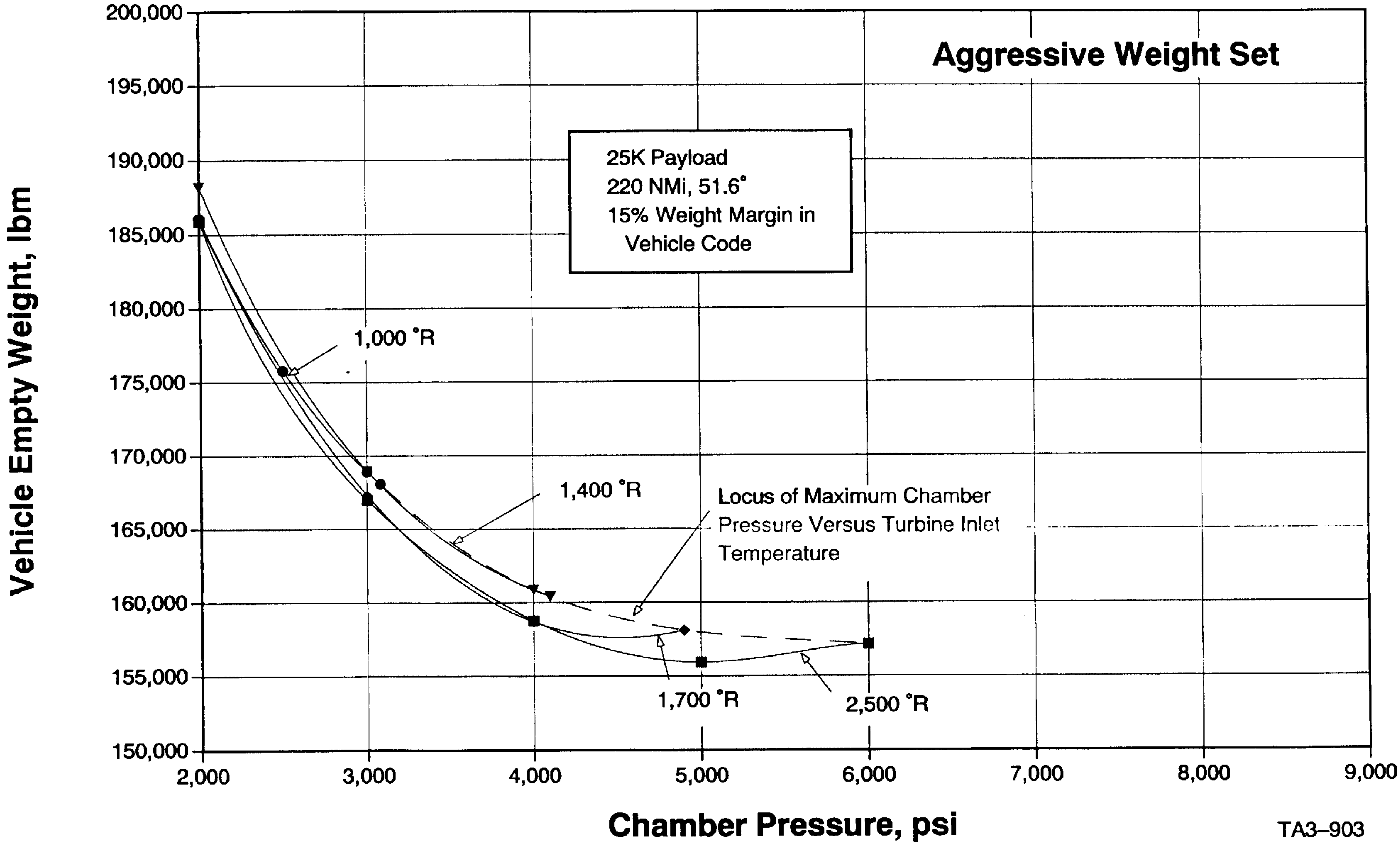


Figure 65. Effect of Fuel Turbine Inlet Temperature on SSTO Performance – Hybrid Cycle

- Based on Turbine Temperature Versus Vehicle Dry Weight Results, Turbine Temperatures Minimized for Each Cycle

- Baseline Engines:

	FFSCC (72b)	SCC (31b)	Hybrid Cycle (12ba)
Eng Weight, lbm	5,003	4,814	4,776
Fuel Tur Temp, °R	1,100	1,200	1,560
Fuel Dis Press, psi	10,670	11,677	10,605
Ox Tur Temp, °R	1,100	1,100	369
Ox Dis Press, psi	9,592	10,984	9,637
SSTO Dry Weight, lbm	162,190	159,264	158,687

Figure 66. Baseline Engine Selection

TA3–0641b

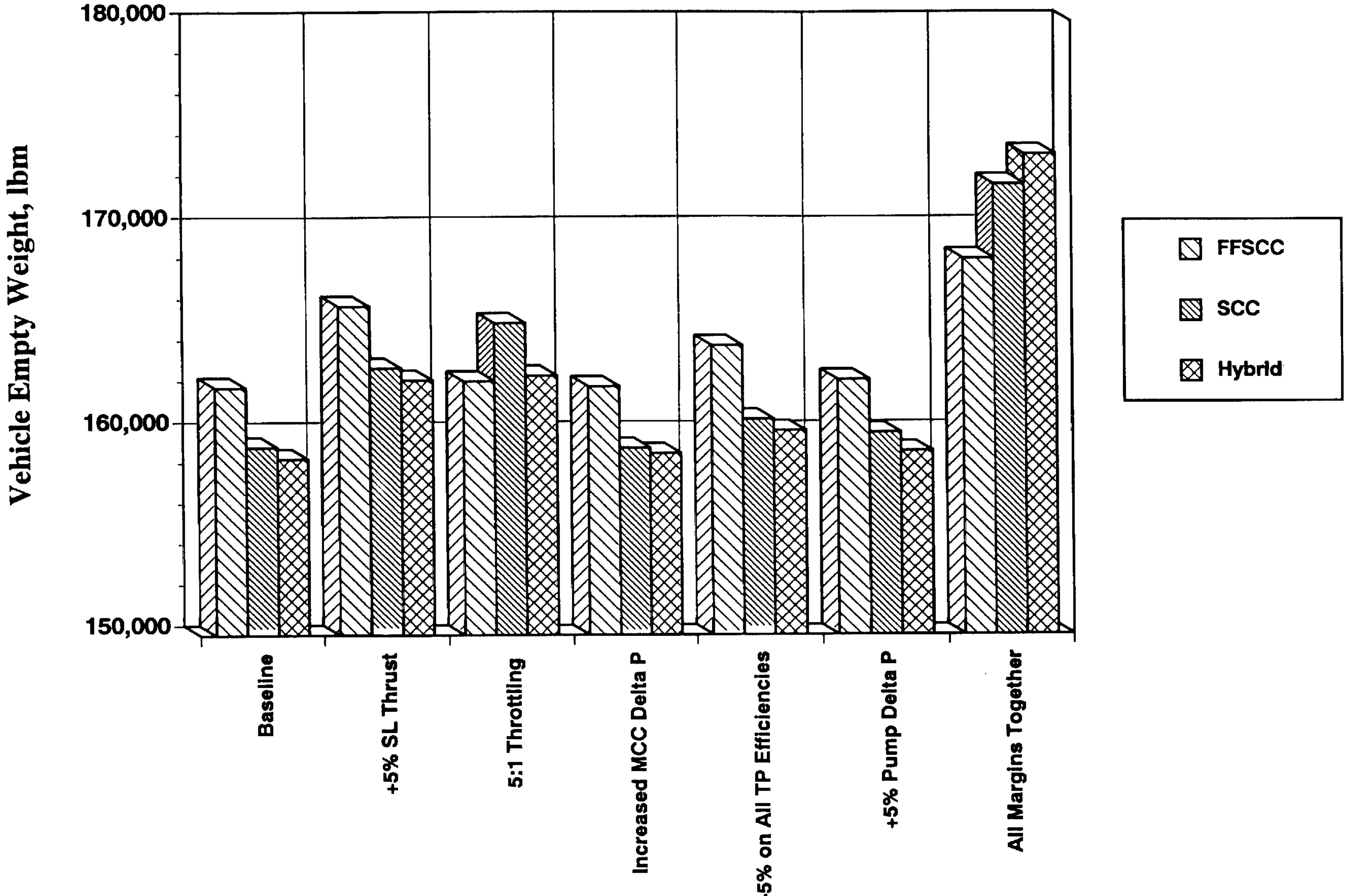


Figure 67. Engine Cycle Sensitivity to Margin Requirements
Pc = 4,000 psi

TA3–0906

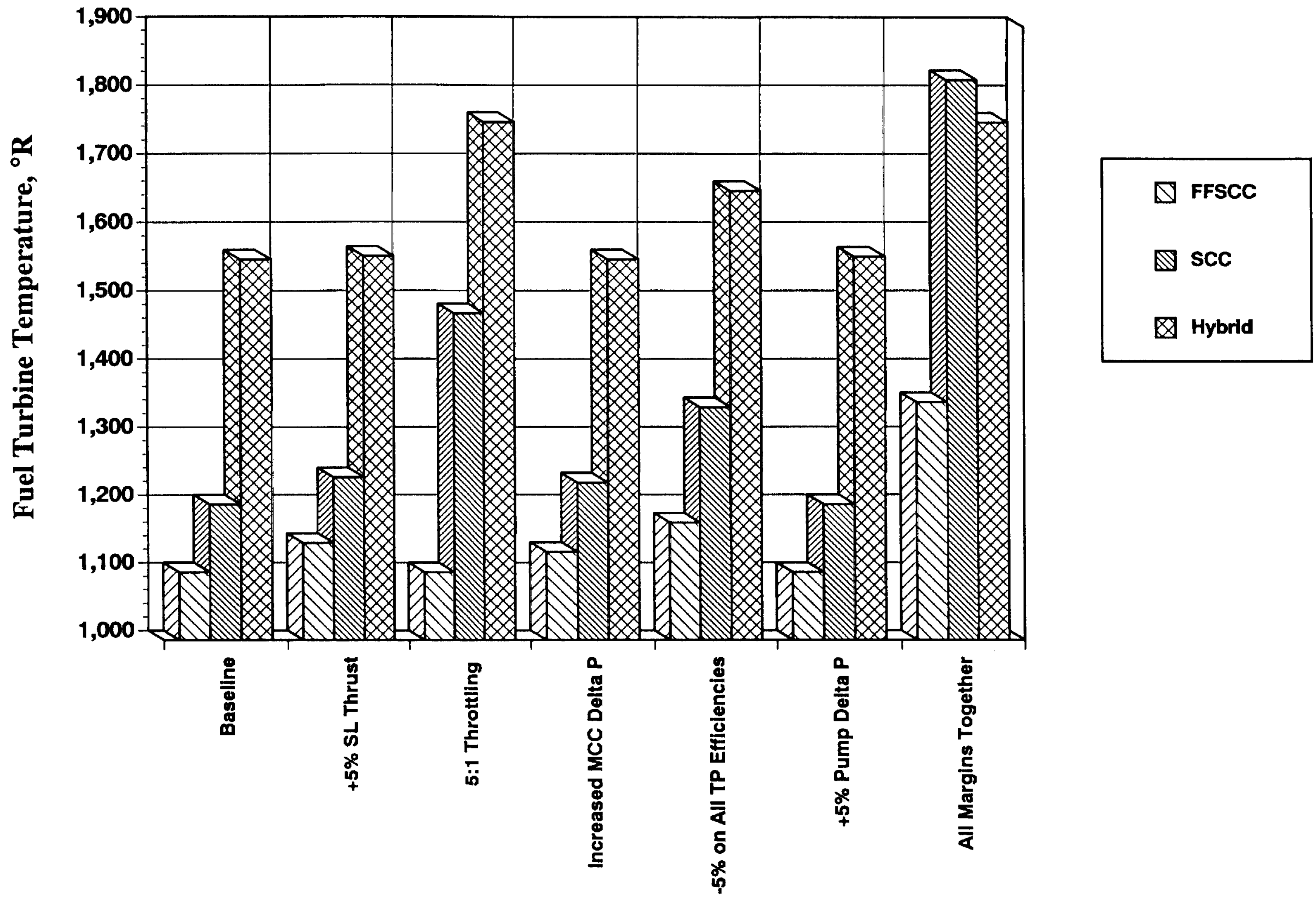


Figure 68. Fuel Turbine Temperature Sensitivity to Margin Requirements
Pc = 4,000 psi

TA3–0907

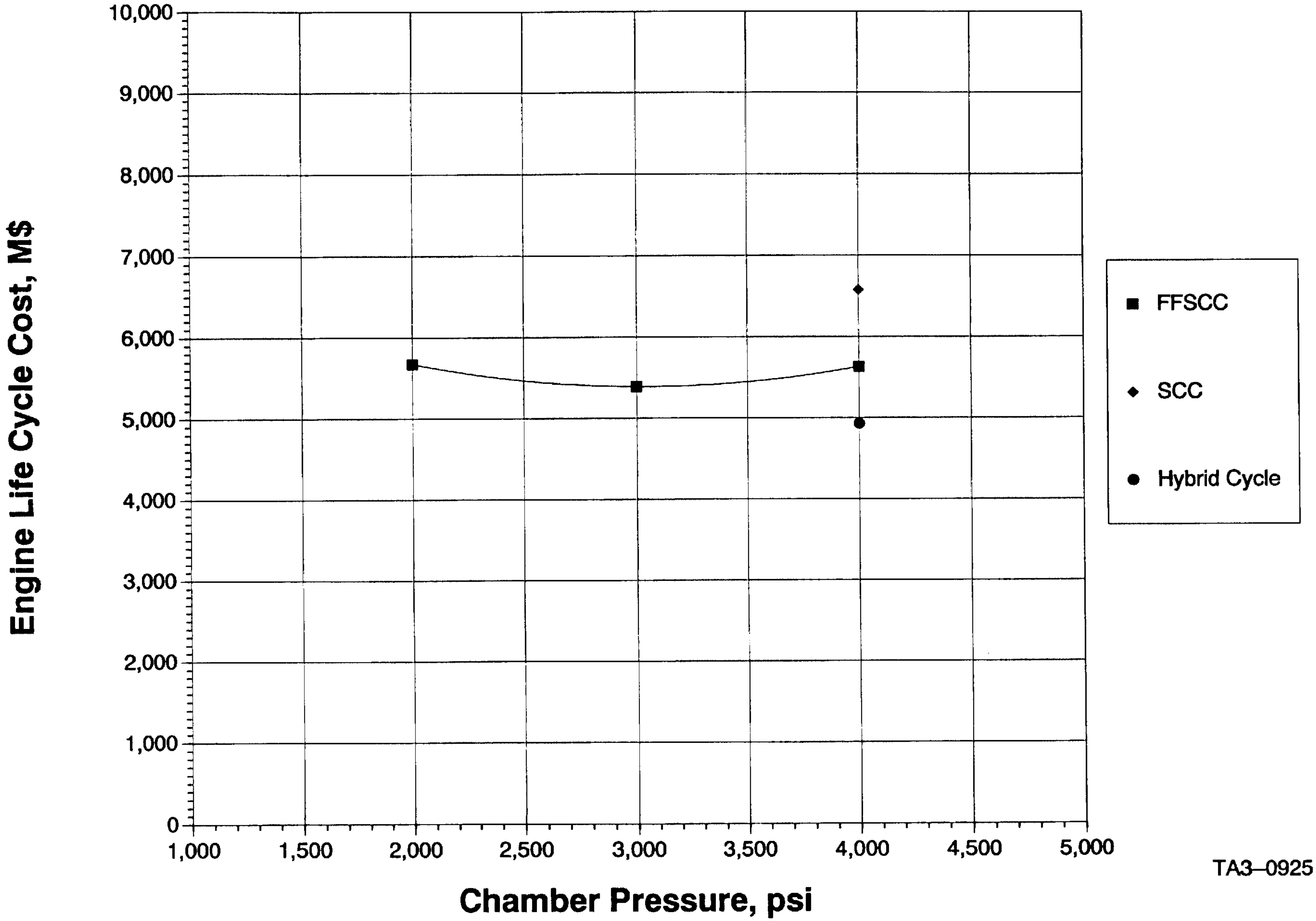


Figure 69. Engine Life Cycle Cost

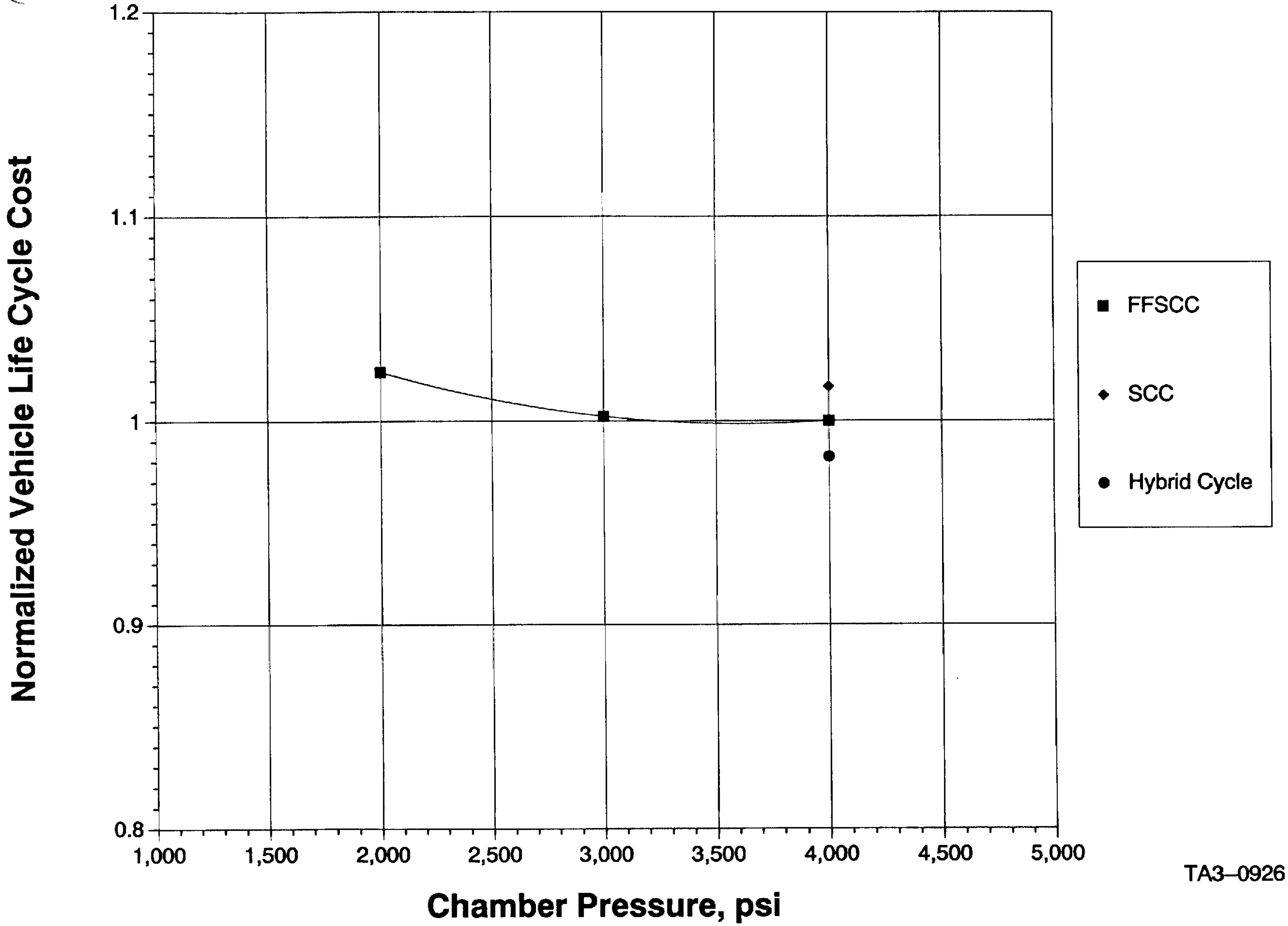


Figure 70. Normalized Vehicle Life Cycle Cost

- Design Point
  - Cycle – FFSCC
  - Chamber Pressure – 4,000 psi
  - Sea Level Thrust – 421,000 lbf
  - Area Ratio – 70.62
  - Fuel Turbine Operating Temperature – 1,100 °R
  - Oxidizer Turbine Operating Temperature – 1,100 °R

- Characteristics
  - Fuel Rich Fuel Turbopump
  - LOX Rich LOX Turbopump
  - Jet Pump Low Pressure Pumps
  - Propellant Duct Gimbal Accommodation on Vehicle Side
  - SLIC™ Turbomachinery
  - Uncooled Powerhead
  - EMA Valves
  - Preburner Injectors Gas/Liq Impinging Jet
  - MCC Injectors Gas/Gas Co-Ax
  - Redundant Laser Igniters
  - Autogenous Pressurization on Both Sides
  - Pump Conditioning Fluid Recirculated to Tank on Both Sides

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Figure 71. Baseline Design Point and Characteristics

- Overall Procedure
  - Various Individual Design Procedures Combined at CATIA Assembly Level for Packaging and in Spreadsheet for Weights

- Two Direct Design Procedures are Used
  - CATIA Solid Model (e.g., Hot Gas Manifold)
    - Designed as Individual Component
    - Wall Thickness Calculated
    - Minimums Applied in Model
      - 1.5 Factor for Dynamic Loads Applied to Wall Thickness if Appropriate
    - Solid Volume Returned to Spreadsheet for Weights
    - In Spreadsheet
      - Density used on Solid Volume for Weight
      - 1.02 Factor and 1.05 Factor Applied to Weight
  - CATIA Assembly Model (e.g., Duct)
    - Designed at Assembly Level for Dimensions, Clearances, and Packaging
    - Dimensions Returned to Spreadsheet for Weights
    - In Spreadsheet
      - Wall Thickness Calculated and Minimums Applied
      - Other Subcomponents Calculated (Flanges, Insulation, Insulation Shields, etc.)
      - Weights Calculated from Material Choices and Dimensions

- Other Procedures are Used For Some Components and May be Combined
  - Scaled (e.g., Valves)
  - Outside Reference (e.g., STME-100 for Controller)
  - Outside Model or Correlation (e.g., SLIC™ Turbomachinery)
  - Directly from SSME (e.g., Static Seals)

Figure 72. Weight Calculation Procedures

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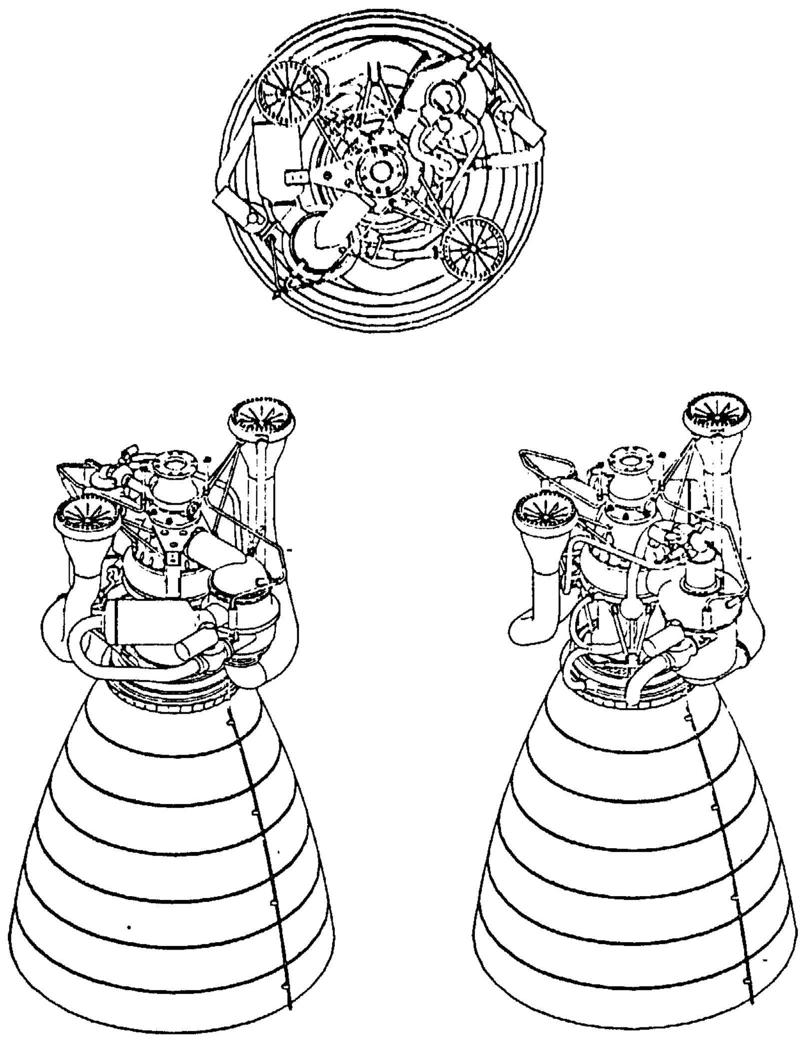
Component Area	SSME Weights, lbm	Adv Low Cost Eng Weights, lbm	Difference lbm	Rationale
Turbomachinery	1,725.00	1,070.01	(654.99)	SLIC™ (387), Jet Pumps (2687)
Nozzle	1,310.54	945.57	(364.97)	Essentially same weight on equal surface area basis (1,342), Ti honeycomb jacket
Hot Gas Manifolds/Inj/ Thrust Cone	953.00	621.70	(331.3)	
Propellant Ducts	822.91	201.38	(621.53)	Gimbal flex accommodation on vehicle side (198), Jet Pump (307), shorter lines and routing
MCC	438.54	450.07	11.53	
Valves	410.62	364.68	(45.94)	Uses EMA Valves. Includes Valves and Actuators
Avionics	375.00	166.74	(207.88)	Controller with FASCOS (221)
Misc	289.30	153.33	(135.97)	Proportional to weight (3.6%)
Preburners	195.75	239.04	43.29	
Gimbal Bearing	105.00	65.63	(39.37)	From Ti to Si carbide reinforced Al
Lines (Interface)	95.32	37.75	(57.57)	Simplified routing, combined recirc and repressurization, less drain
Pneumatics	76.90	0	(76.90)	EMA valves
POGO	75.13	40.41	(34.72)	Stiffer System, 25% SSME gas
Hydraulics	30.32	0	(30.32)	EMA Valves
Heat Exchanger	26.00	0	(26.00)	Part of LOX rich preburner
Igniters	26.00	6.00	(20.00)	Laser Igniters
Purge	24.39	24.39	0	Left in for ground Ops
Bleed Recirc Pumps	10.00	20.00	10.00	Add to LOX side
Static Seals	6.00	6.00	0	
	6,995.72	4,412.70	(2,583.02)	

Figure 73. Weight Comparison to SSME

TA3–0632



Full Flow Staged Combustion Cycle
(FFSCC) Mixed Preburners
Bipropellant LOX/H_2

Chamber Pressure, psi	4,000
Thrust, lbf	421,000 (SL) 486,867 (Vacuum)
Area Ratio	70.62
I_{sp}	395.4 (SL) 457.2 (Vacuum)
Weight, lbm	4,413
Diameter, in	84
Length, in	147

Figure 74. Advanced Low-Cost Engine Study Baseline Engine Configuration

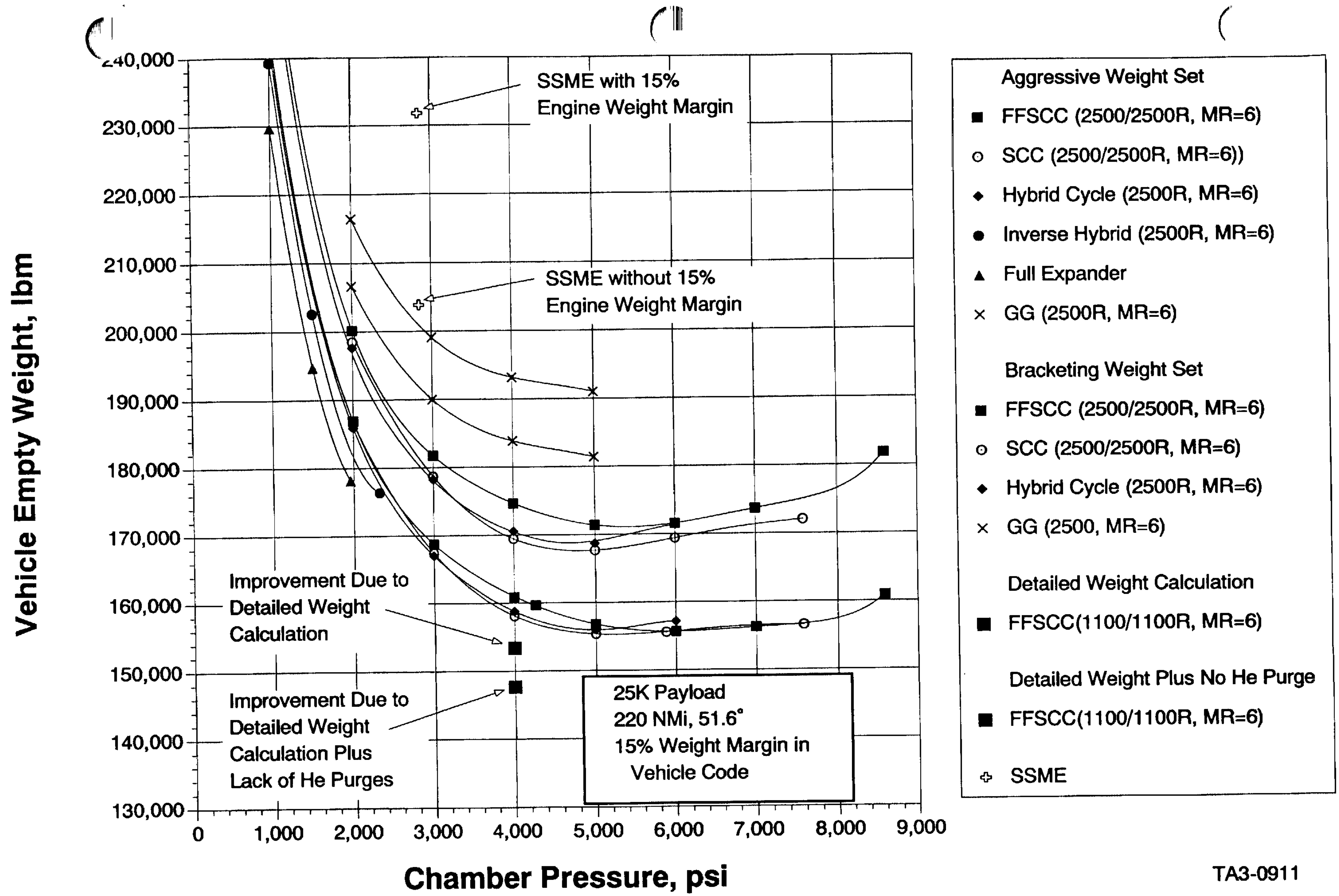
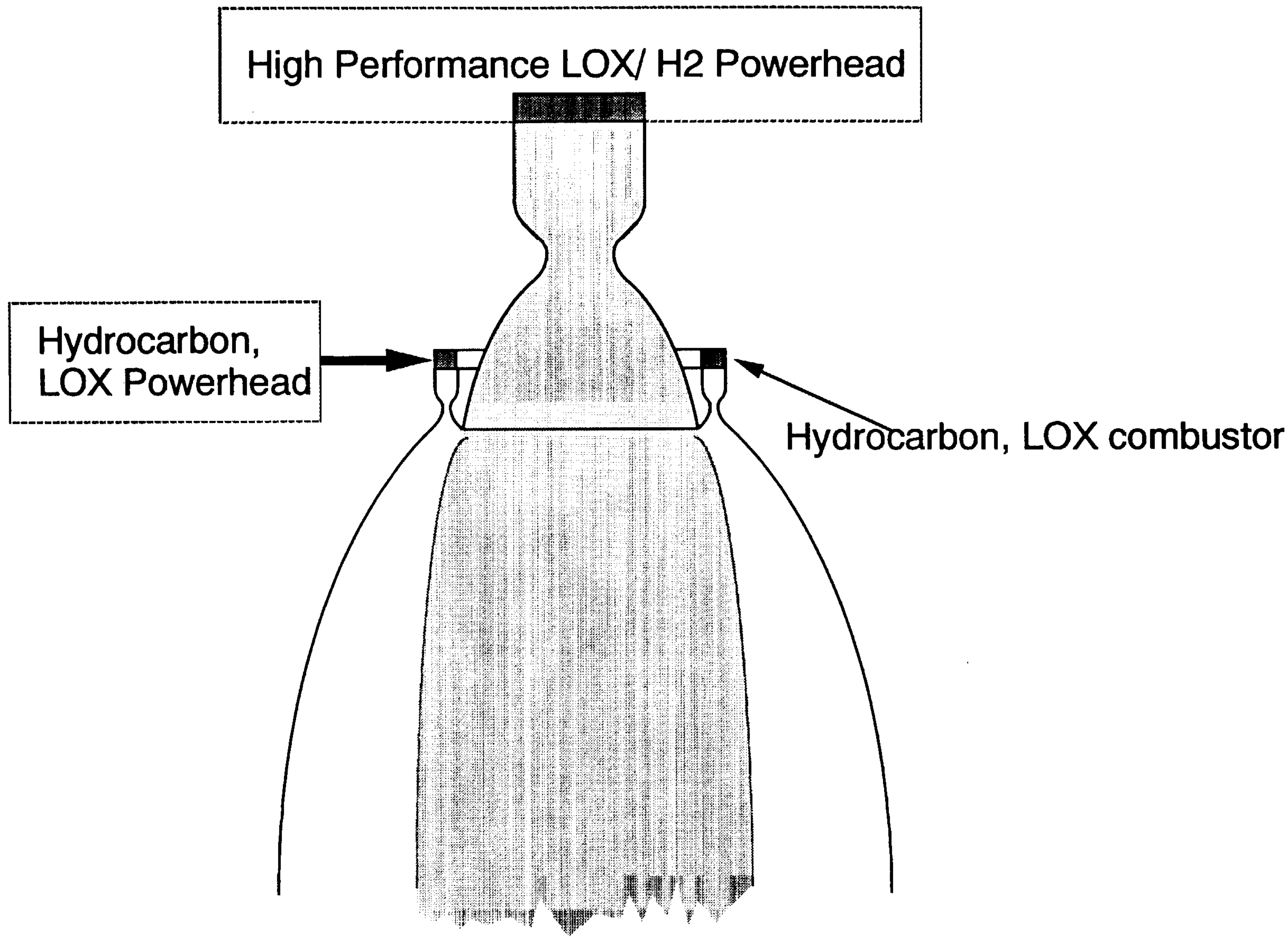


Figure 75. Advanced Low-Cost Engines SSTO Performance



TA3–0539c

Figure 76. Bell Annular Tripropellant Concept

Parameter	Case				
Configuration	Bipropellant Fixed Bell	Tripropellant Bell Annular	Tripropellant Bell Annular	Tripropellant Bell Annular	Tripropellant Bell Annular
Propellants	Lox/H_2	$LOX/H_2/RP$	$LOX/H_2/RP$	$LOX/H_2/RP$	$LOX/H_2/RP$
Cycle	FFSCC	FFSCC (LOX/H_2) GG (LOX/RP)	FFSCC (LOX/H_2) GG (LOX/RP)	FFSCC	FFSCC
Mode 1					
P_C, psi	4,000	3,000	4,000	3,000	4,000
T_{sl}, lbf	421,000	421,000	421,000	421,000	421,000
T_V, lbf	486,867	474,422	470,012	475,724	471,071
I_{spsl}, sec	395.4	315.0	318.9	322.0	328.8
I_{spv}, sec	457.2	354.9	356.1	363.8	368.0
Area Ratio	70.62	44	55	44	55
Mode 2					
P_C, psi	–	3,000	4,000	3,000	4,000
T_{sl}, lbf	–	–	–	–	–
T_V, lbf	–	144,780	142,875	143,560	142,785
I_{spsl}, sec	–	298.7	313.6	289.0	304.9
I_{spv}, sec	–	463.4	466.4	464.2	467.1
Area Ratio	–	141	174	150	185
Engine Weight, lbm	4,413	4,318	4,271	5,020	4,690
Engine SL T/W	95.4	97.5	98.6	83.9	89.8

Figure 77. Case Parameters

TA3–0662

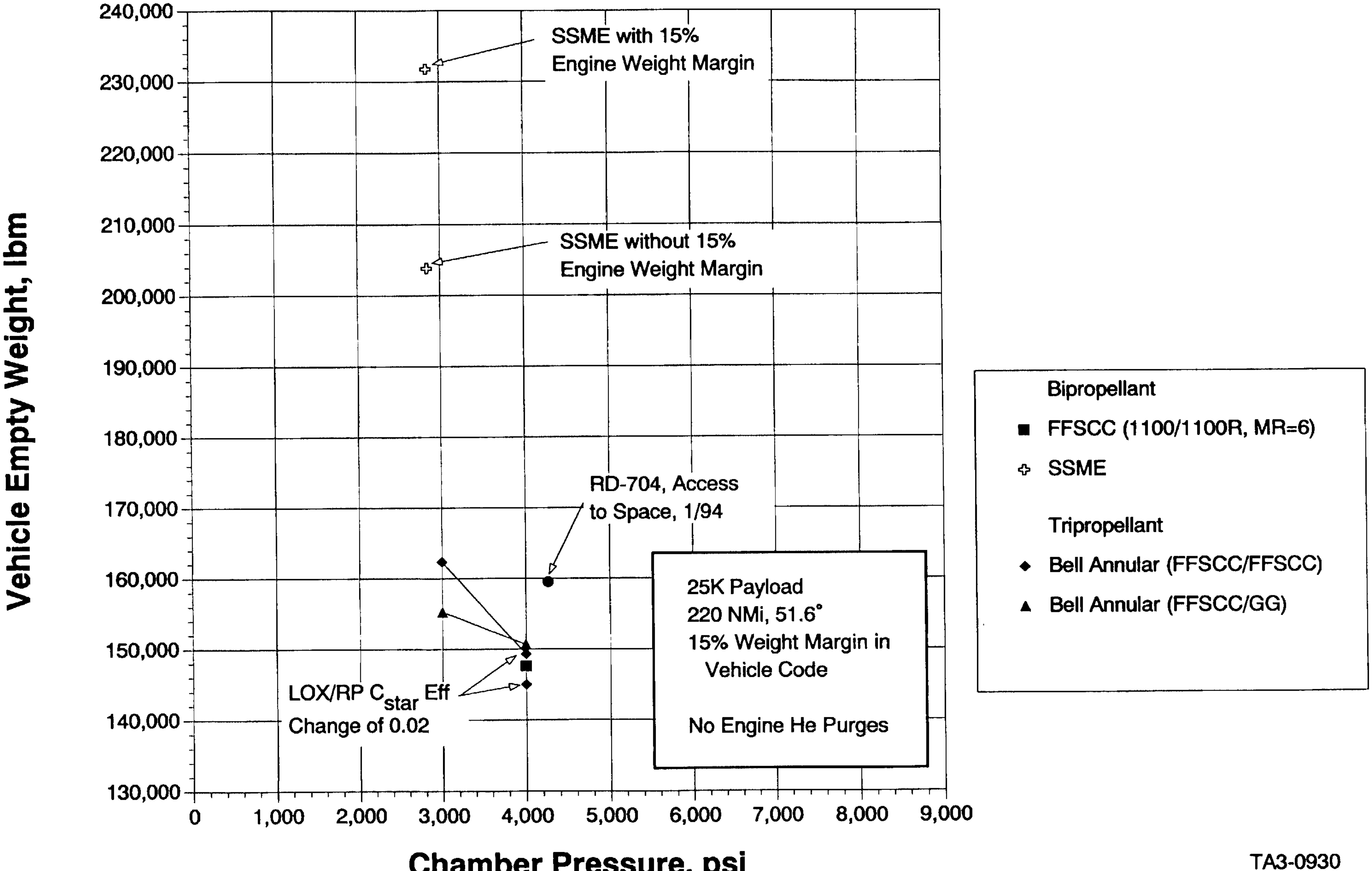


Figure 78. SSTO Performance Bipropellant vs Tripropellant

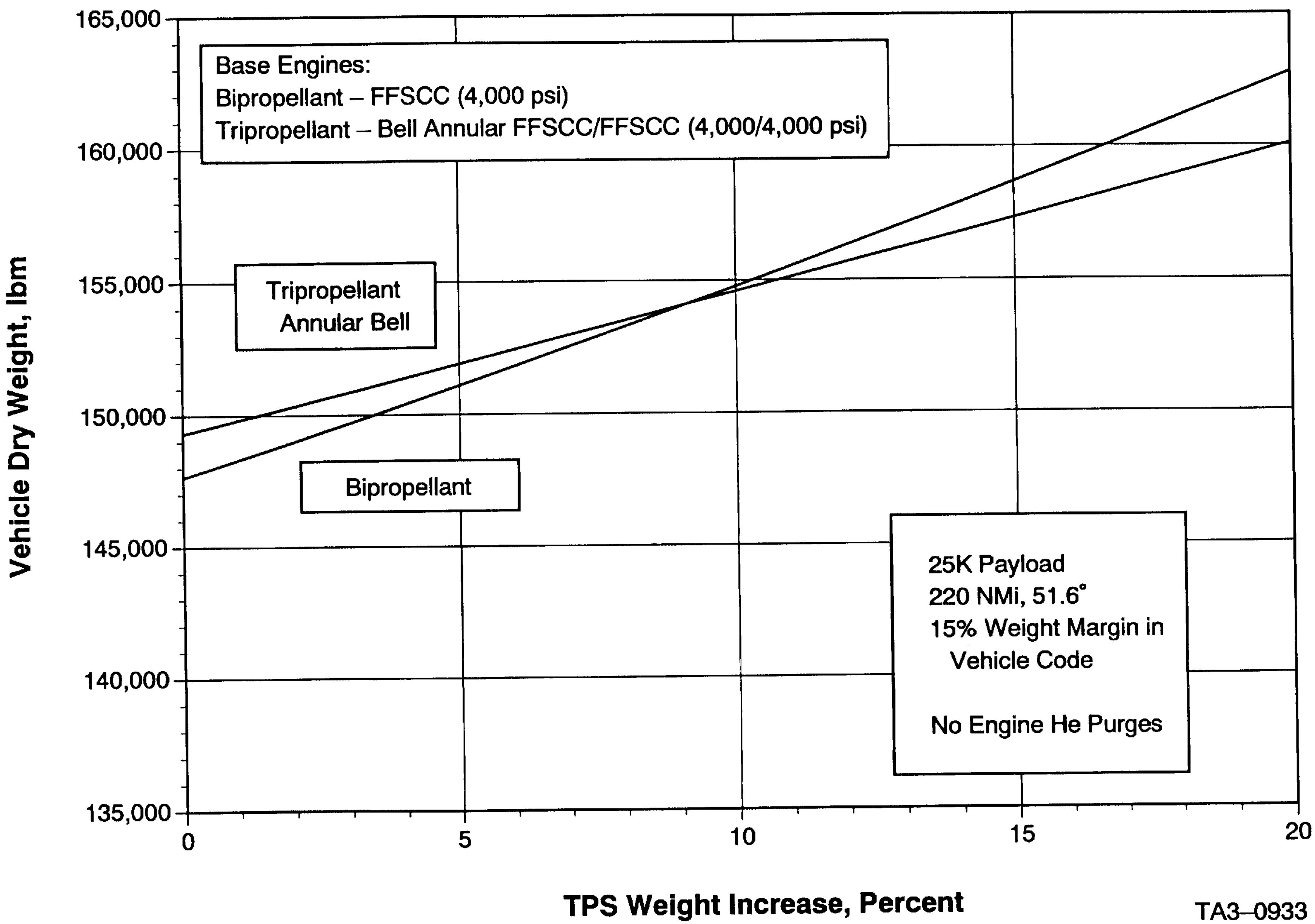


Figure 79. Effect of TPS Weight Growth

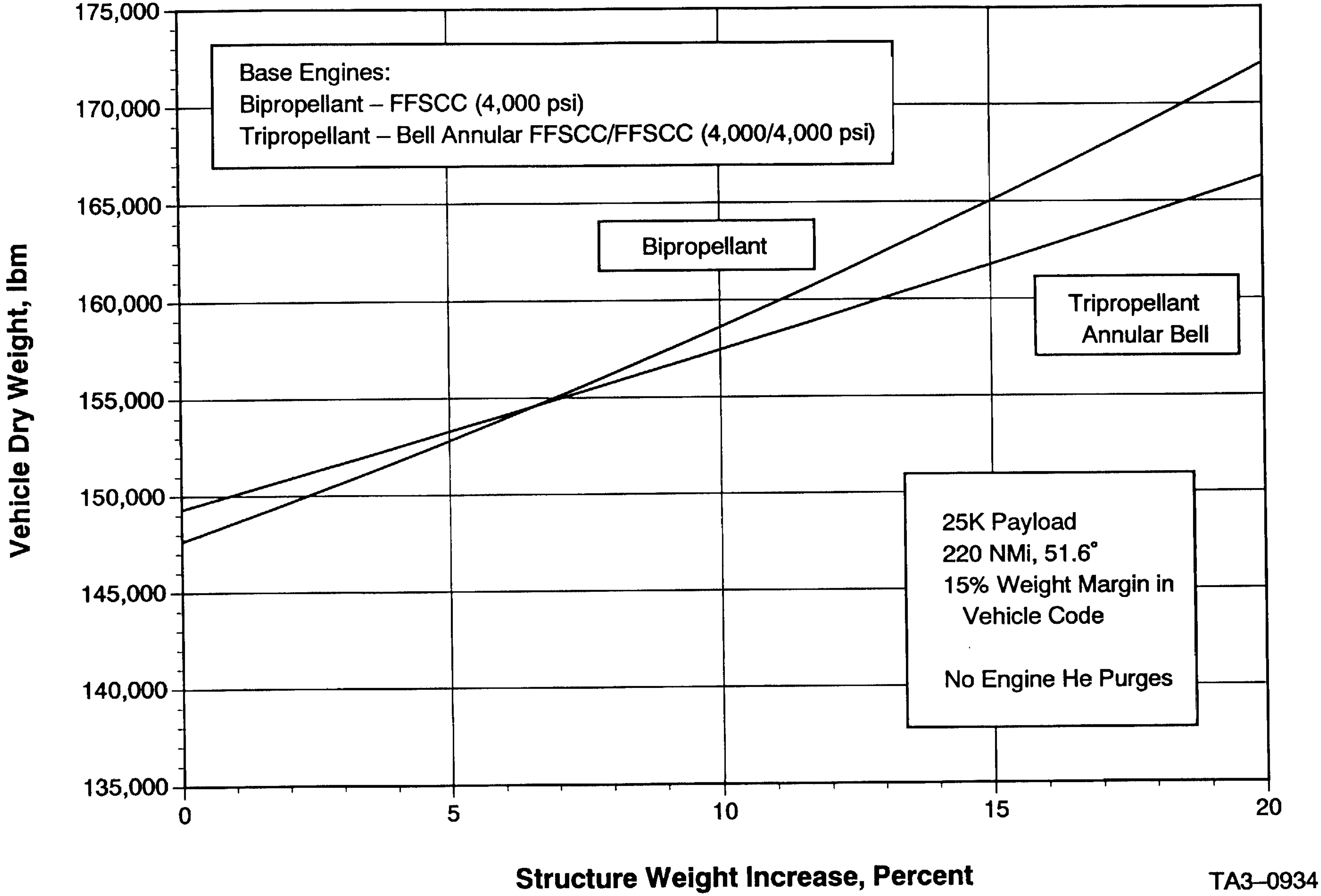
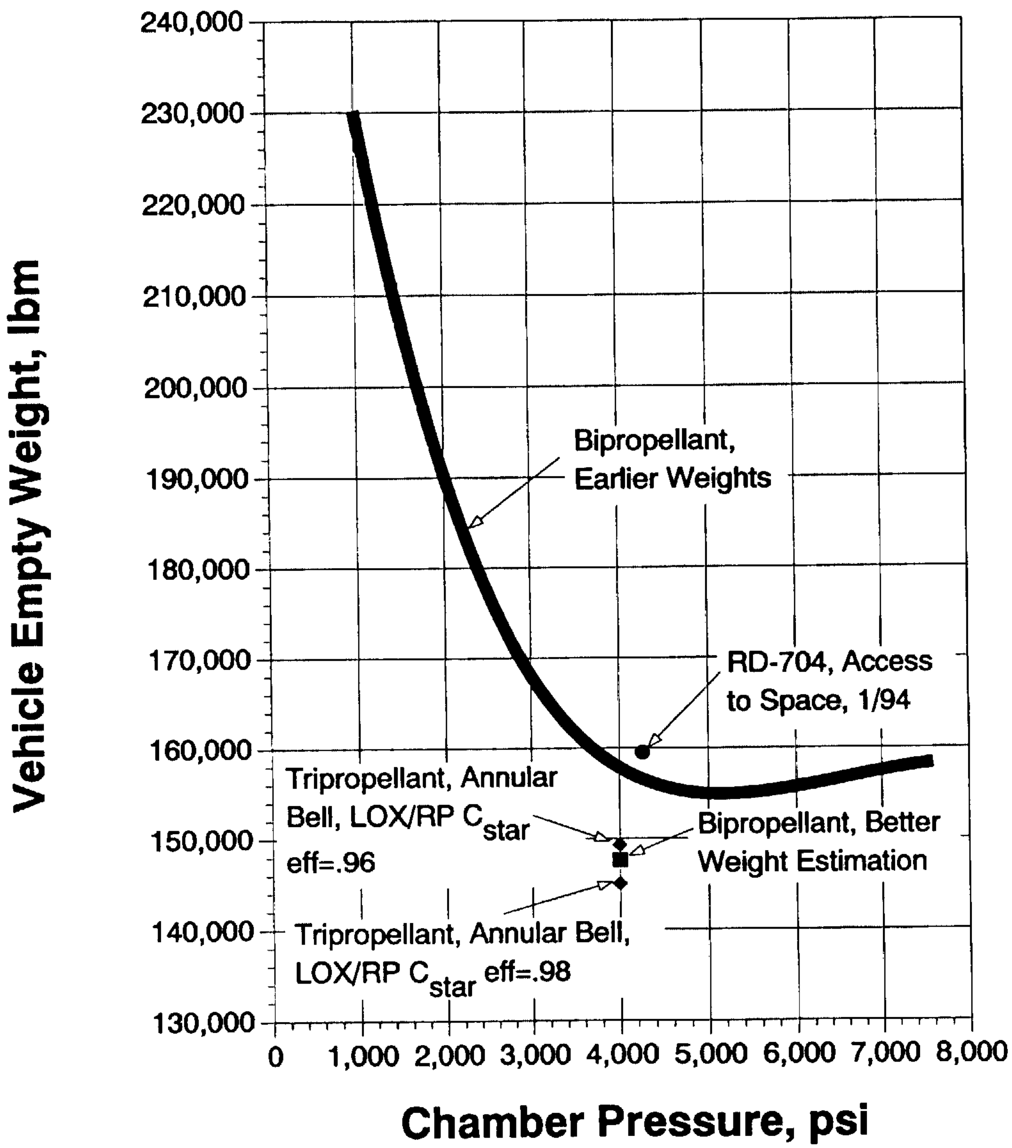


Figure 80. Effect of Structure Weight Growth



- **Conclusions**
  - **SSTO is Reasonable Based on Near to Mid Term Engine Technologies**
  - **Bipropellant and Tripropellant Approaches Both Viable**
  - **Only Moderately High P_c Needed**
    - **~ 4,000 psi**
  - **Needed Turbine Temperatures are Low**
    - **Fuel 1,000 - 1,700 °R**
    - **LOX 1,100 - 1,300 °R**
  - **Allows High P_c With Durability**
    - **Low Turbine Temperatures**
  - **Cost Reduction Opportunities**
    - **Simplified Turbomachinery**
    - **Jet Pumps**
    - **Low Turbine Temperatures**
    - **Material Choices**

TA3–0935

Figure 81. Advanced Low-Cost Engines SSTO Results Summary

Technology Areas	Bi-Propellant	Tripropellant	Impact	Increase in Vehicle Dry Weight if Not Used
Increased P_C	X	X	Significant Weight Reductions Up to ~ 4,000 psi	
Lower Turbine Operating Temperatures	X	X	Margin, Ops Costs	
LOX Rich LOX Turbopumps	X	X	Margin, Ops Costs Thru Lower Turbine Temperatures by Allowing Cycles Which are Less Sensitive in Turbine Operating Temperature versus ΔP, Throttling, and P_C	
LOX Rich Preburners	X	X		
SLIC™ Turbomachinery	X	X	Significant Weight Reductions, Better Ops	+7.5%
Jet Pumps	X	X	Significant Weight Reductions, Better Ops, Lower Costs	+5.8%
Vehicle Side Gimbal Flex Accommodation	X	X	Significant Weight Reductions on Engine	+1.9%
Al Fuel Pump	X	X	Lower Turbomachinery Weights	+1.2%
Laser Ignition	X	X	Easier Development, Better Ops	
Gasify LOX	X	X	Margin for Deep Throttling (e.g., 5:1)	
Health Monitoring/Life Prediction	X	X	Reliability, Ops Costs	

Figure 82. Advanced Low-Cost Engine Study Technology Implications

TA3–0587d

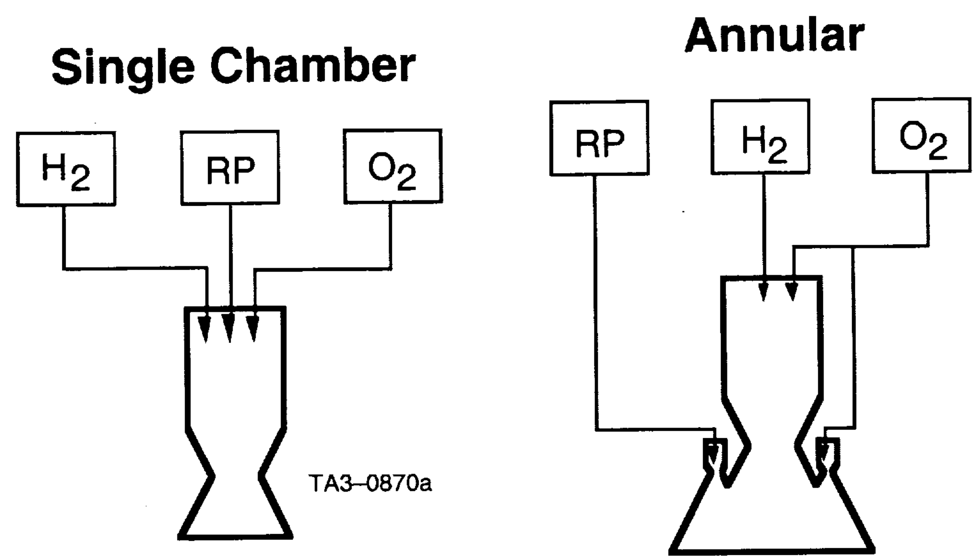


Figure 83. Tripropellant Configurations

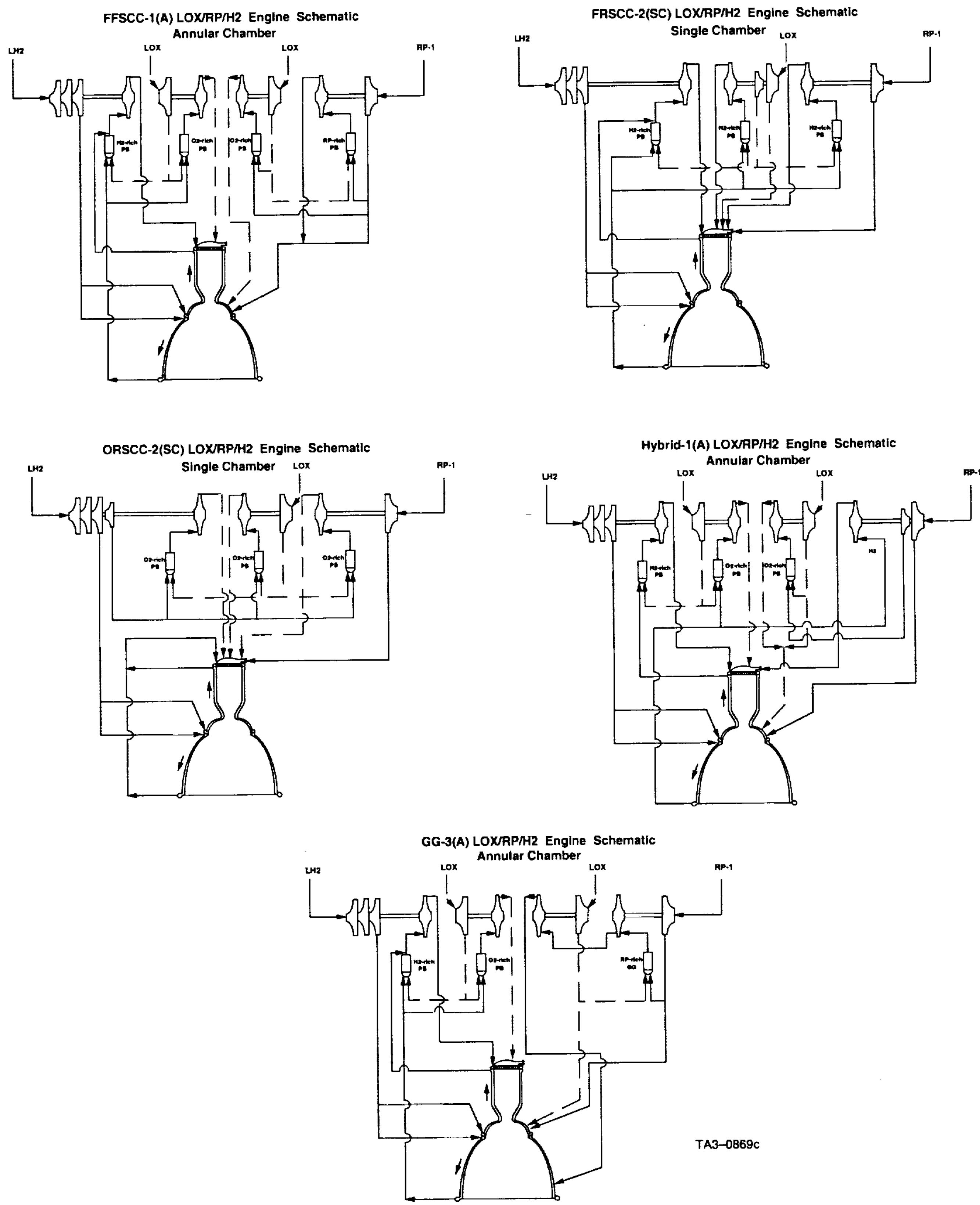


Figure 84. Representative Engine Cycles

	Single Chamber Tripropellant	Bell Annular Tripropellant	Bipropellant Closed Cycles	Bipropellant Gas Generator
Thrust, Sea Level, lbf	421,000	421,000	421,000	421,000
Thrust, Vacuum, lbf	477,630	478,701	484,585	486,706
Specific Impulse, sec				
Mode 2 Vacuum	450.69	461.13	451.43	445.28
Mode 2 Sea Level	339.18	267.33	392.19	385.16
Mode 1 Vacuum	406.26	369.33	451.43	445.28
Mode 1 Sea Level	358.09	324.81	392.19	385.16
Chamber Pressure, psi				
Mode 1	4,000	4,000	4,000	4,000
Mode 2	1,966	4,000	4,000	4,000
Area Ratio				
Mode 1	63.56	59.60/64.54*	69.77	69.84
Mode 2	63.56	226.73	69.77	69.84
Engine Weight, lbm				
FFSCC	4,176	4,201	4,242	—
ORSCC	4,295	—	—	—
FRSCC	4,040	4,187	4,049	—
Hybrid Cycle	4,026	4,227	4,058	—
Gas Generator Cycle	—	—	—	3,629

TA3–1129

* $(O_2/H_2)/(O_2/RP)$

Figure 85. Engine Characteristics

	Single Chamber Tripropellant	Annular Tripropellant	Bipropellant
Nozzle Exit Pressure, psi	6.0	5.5	4.5
Mode 1 Mixture Ratio	4.4	—	—
Mode 2 Mixture Ratio	6.2	—	—
O_2/RP Mixture Ratio	—	2.8	—
O_2/H_2 Mixture Ratio	—	6.8	6.9
Percent Hydrogen, %	6	—	—
Mode 1 O_2/RP to O_2/H_2 Thrust Split	—	H_2 Cooling Limit	—

TA3–0878c

Figure 86. Baseline Parameter Selections

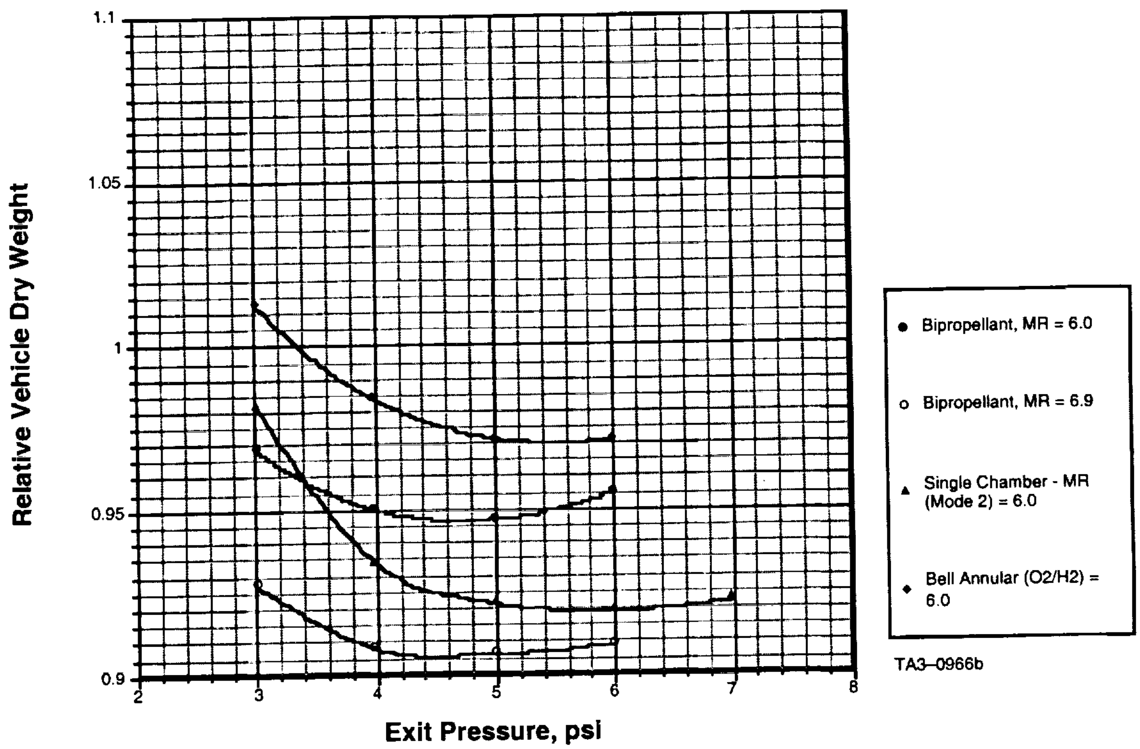


Figure 87. Nozzle Exit Pressure Optimization

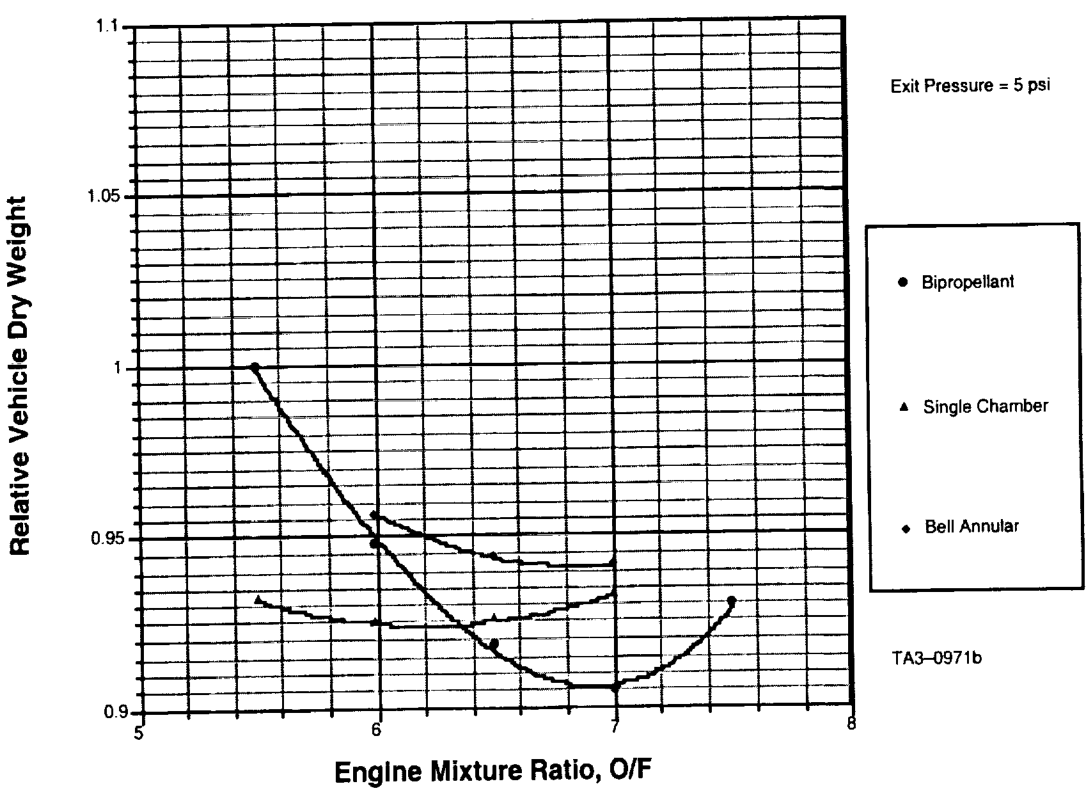


Figure 88. Engine O_2/H_2 Mixture Ratio Optimization

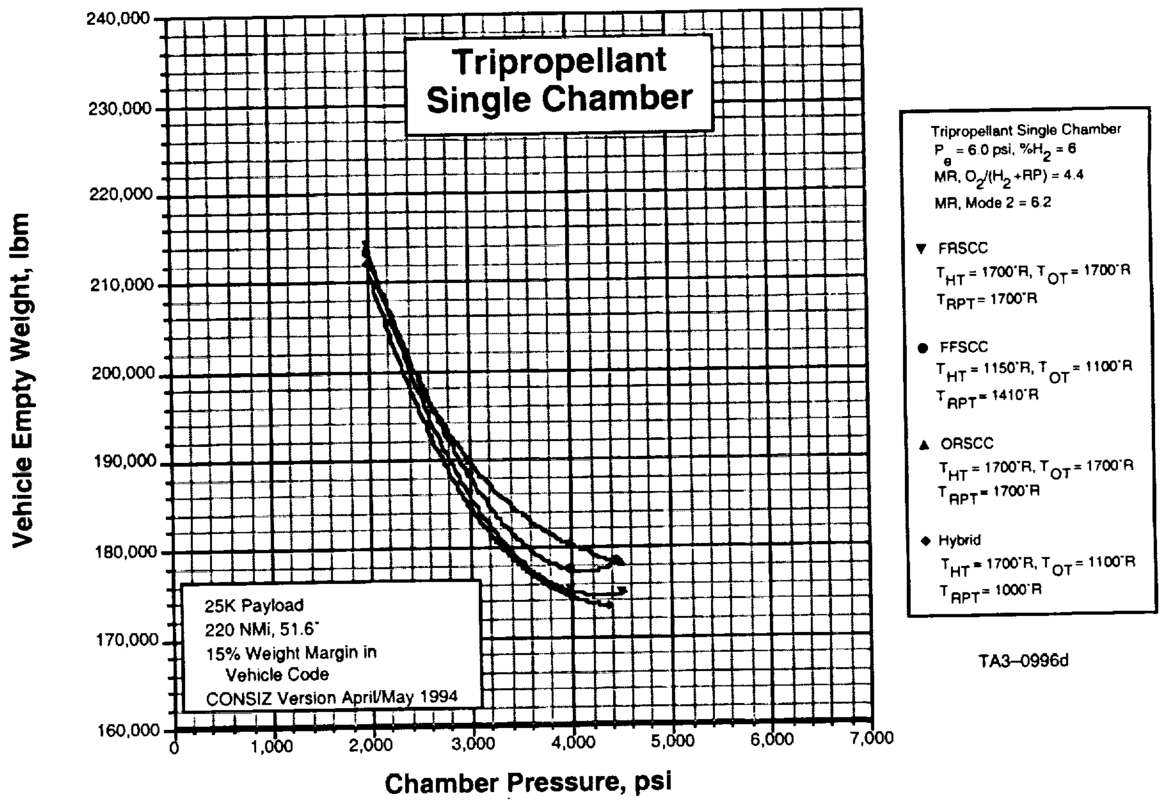


Figure 89. Single Chamber Tripropellant Vehicle Performance

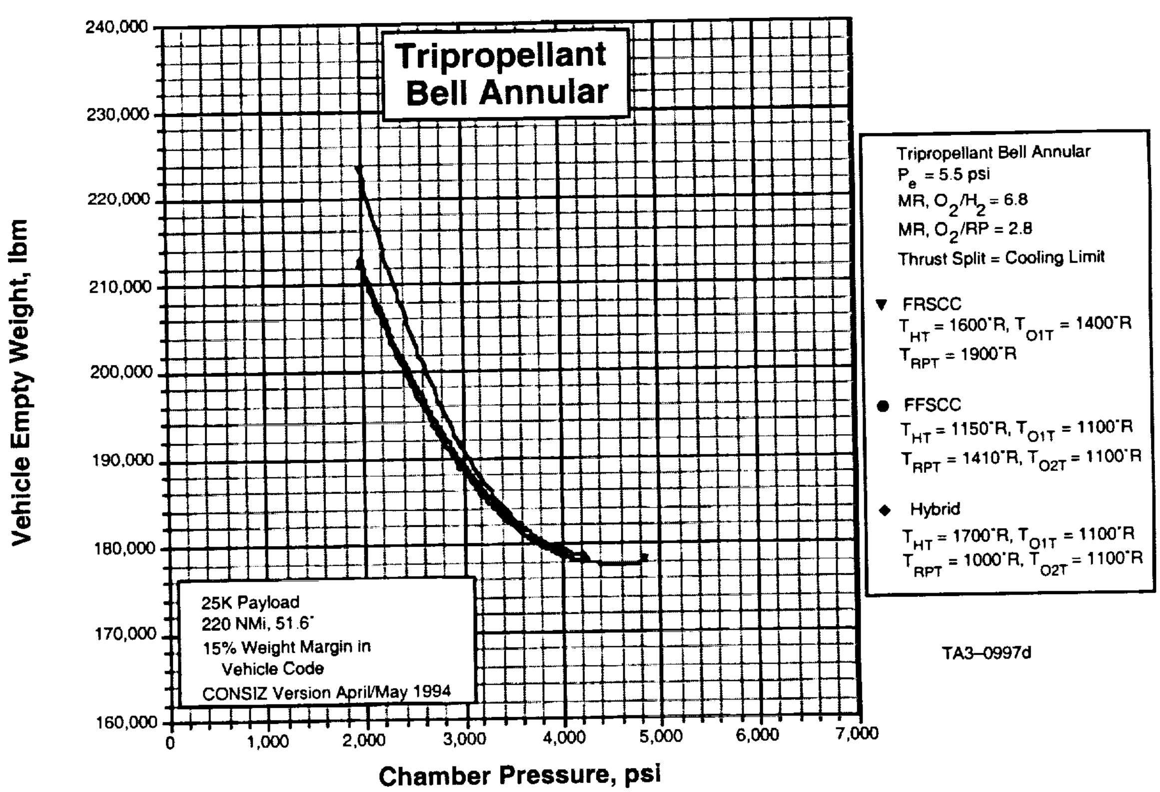


Figure 90. Bell Annular Tripropellant Vehicle Performance

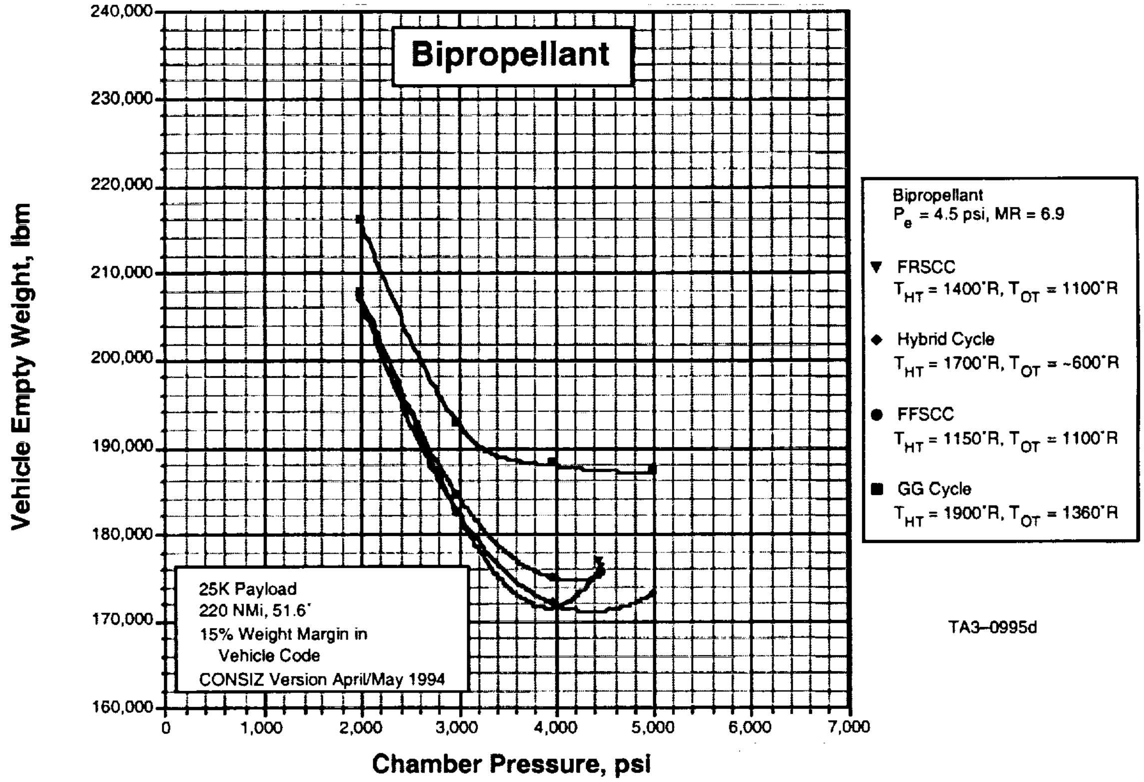


Figure 91. Bipropellant Vehicle Performance

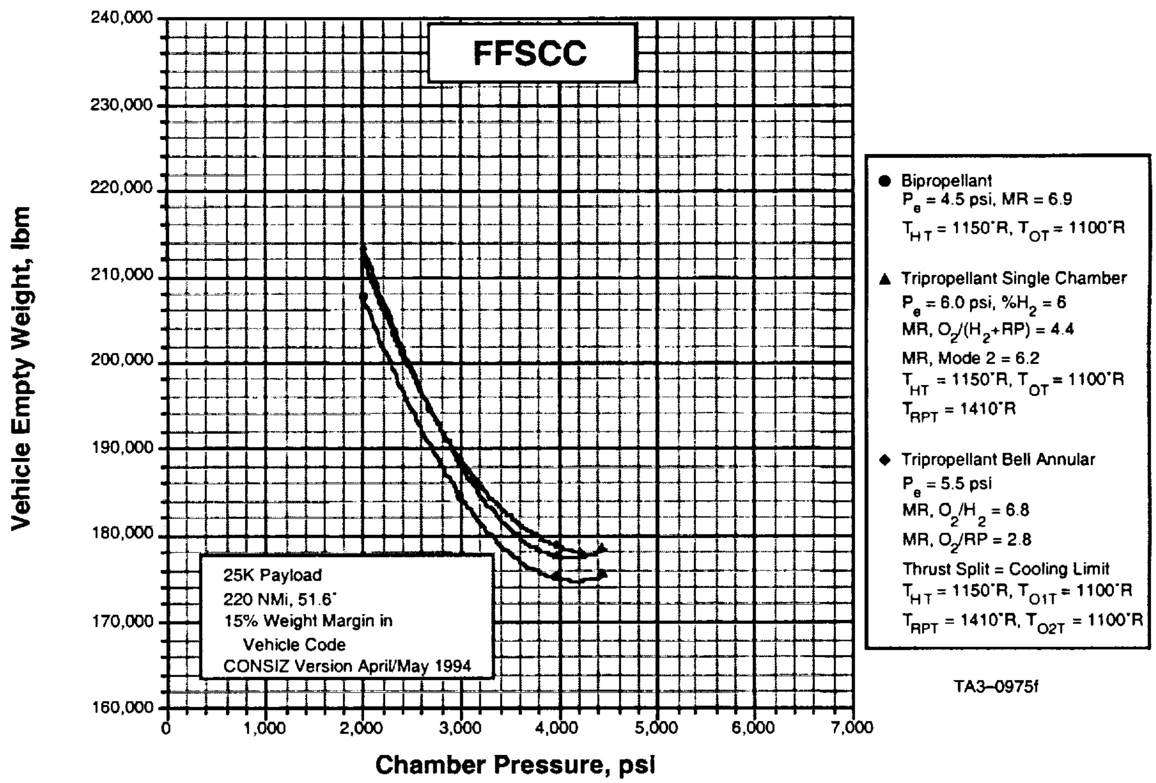


Figure 92. Tripropellant Versus Bipropellant Vehicle Performance